150p.

N64-17690
CODE-1
CR-55931

[redacted]

OTS PRICE

XEROX $ 12.50 ph
MICROFILM $ 5.60 mf.

REPUBLIC
AVIATION CORPORATION

RAC-1333; Vol. 2;
~~(SSD-1027)~~ SSD-1027, vol. 2)
open auth. 17 June 1963
1808 refs

SYNCHRONOUS METEOROLOGICAL SATELLITE
(SMS) STUDY.

Volume 2:

Configurations and Systems 7@ [Final Report, Feb.-May 1963]

~~RAC-1333~~

OTS: $12.50ph, $5.60mf

Prepared for
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Goddard Space Flight Center
Greenbelt, Maryland

under
(NASA Contract ~~No.~~ NAS5-3189)

1948322

REPUBLIC AVIATION CORPORATION
Farmingdale, N.Y.

FOREWORD

This final report on Contract NAS 5-3189 is presented by Republic Aviation Corporation to the Goddard Space Flight Center of the National Aeronautics and Space Administration and consists of the seven volumes listed below. The period of the contract work was February through May, 1963.

The sub-titles of the seven volumes of this report are:

1 Summary and Conclusions

2 Configurations and Systems

3 Meteorological Sensors

4 Attitude and Station Control

5 Communications, Power Supply, and Thermal Control

6 System Synthesis and Evaluation

7 Classified Supplement on Sensors and Control

Except for Volume 7, all of these are unclassified. Volume 7 contains only that information on specific subsystems which had to be separated from the other material because of its present security classification. Some of these items may later be cleared for use in unclassified systems.

Volumes 3, 4, and 5 present detailed surveys and analyses of subsystems and related technical problems as indicated by their titles.

In Volume 2, several combinations of subsystems are reviewed as complete spacecraft systems, including required structure and integration. These combinations were selected primarily as examples of systems feasible within different mass limits, and are associated with the boosters to be available.

Volume 6 outlines methods and procedures for synthesizing and evaluating system combinations which are in addition to those presented in Volume 2.

Volume 1 presents an overall summary and the principal conclusions of the study.

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SECTION 1-SYSTEM CONCEPTS AND OBJECTIVES

A. INTRODUCTION

As stated in Contract NAS5-3189, the purposes of the investigations of synchronous meteorological satellite (SMS) systems conducted by Republic are:

(1) To find the most reasonable and reliable systems for bringing maximum areas of the Earth under constant observation.

(2) To identify the critical scientific and engineering problem areas and the advances in technology required.

(3) To review possible system trade-offs and alternatives, primarily against effects on quality of meteorological data outputs.

An SMS system has been proposed for inclusion in a long range plan for a global meteorological satellite system. Tiros satellites have demonstrated both the feasibility and the utility of observing cloud cover and other meteorological phenomena over large areas from a spacecraft. This type of observation has provided substantial advantages by supplementing the older types of observation limited to the Earth's surface and atmosphere. Planned as a successor to Tiros is the Nimbus satellite system, which will have near-polar orbits. This satellite is now well along in development, with first launch expected in the near future. The addition of the SMS system, which will have a synchronous equatorial orbit is planned to begin some time after 1966. Some of the advantages to be gained by the addition of the SMS are:

(1) Expanded area of simultaneous and continuous coverage-about a third of the globe for one satellite.

(2) More frequent coverage-48 times per day is specified in the contract.

(3) Broadcast delivery of data, immediately after observation, direct to any desired number of user locations within the coverage area; hence, reduced delays, "fresher" data, and reduced requirements for surface communications facilities.

These advantages and others stem from the synchronous equatorial orbit planned for the SMS. At its altitude of 22,240 miles, the satellite will remain over one spot on the equator. The contract specifies this location as latitude 0, longitude 90°W, which is near the Galapagos Islands, west of South America, and roughly due south from New Orleans and the Yucatan Peninsula of Mexico. The satellite views the near hemisphere defined by latitudes 80°N and S and longitudes 10 and 170°W.

The principal elements, links, functions, and types of data to be delivered, as specified by the contract, are summarized in Figure 1-1.

The meteorological data indicated in this figure are defined in the contract as cloud cover over the viewable disc of the Earth, to be obtained at least once every 30 minutes through each 24 hour day, plus data for assessment of details

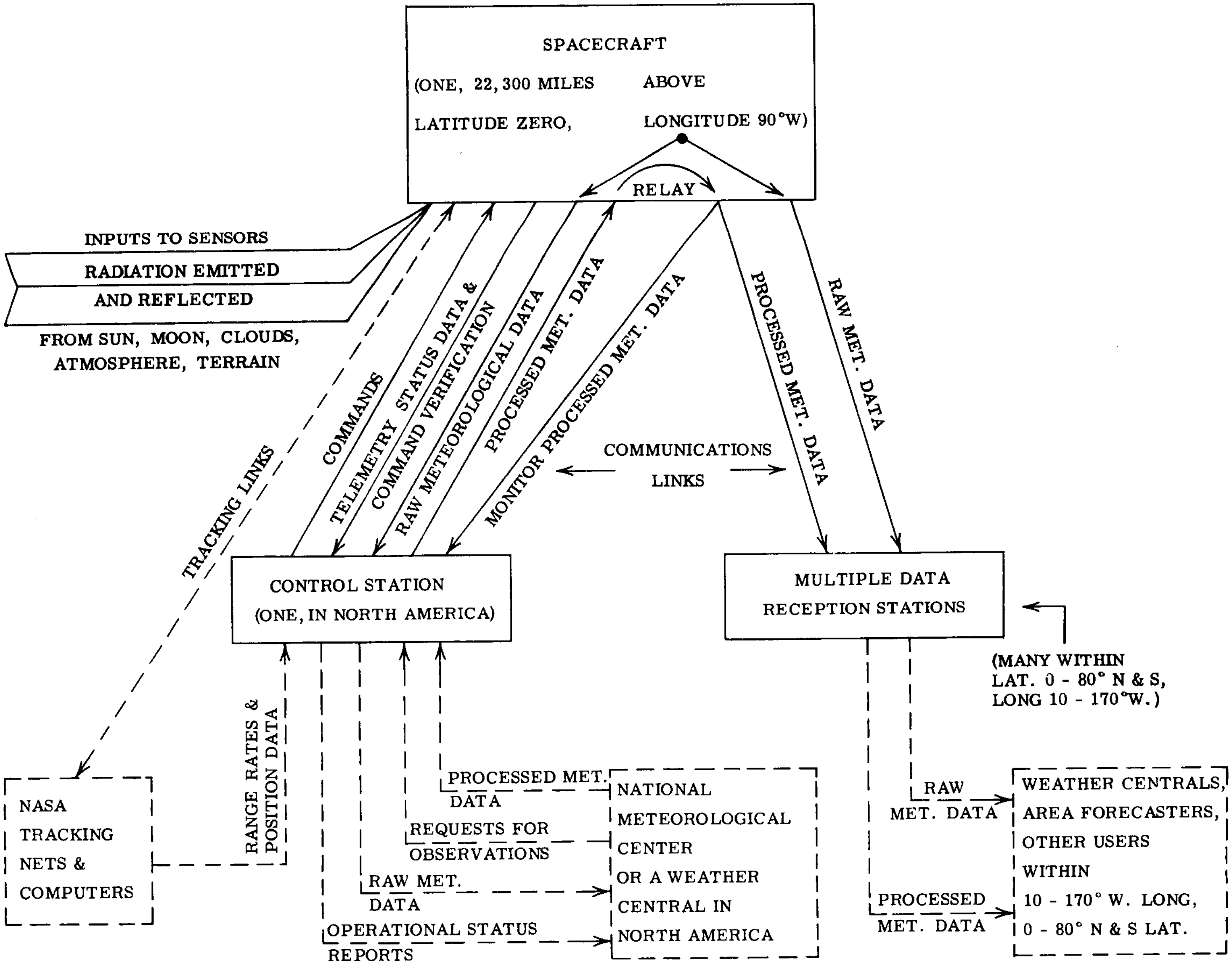

Figure 1-1. SMS System - Principal Elements and Links (Broken Lines Indicate Items External to SMS System)

of cloud systems in selected smaller areas, and infrared measurement of the Earth's heat budget. Figure 1-2 indicates the types of radiations with which the meteorological sensors must work. Figure 1-3 illustrates the principal functions and relationships for a sensor channel in the SMS. The nature of the "processed meteorological data" was not specified in the contract; personnel of NASA have indicated that the principal type to be considered is line drawing types of material, such as weather maps and nephanalysis charts. Other forms of data to be transmitted are also considered in Volume 5.

Two situations are to be recognized for the operations of the SMS system: the R&D phase, and the operational phase. This study has been directed mainly toward development of the operational system, as indicated in Figure 1-1.

In the operational phase, as well as the R&D phase, it is of primary importance to receive the satellite data at the Control and Data Acquisition (CDA) Station. The utility of the synchronous meteorological satellite will be enhanced by providing for the transmission of its sensor data to numerous ground locations, referred to as Multiple Data Acqusition (MDA) stations. The number of MDA stations will in some measure depend upon the cost of such installations. The position of the single spacecraft studied for this report is 90° W longitude over the equator, which is just off the west coast of South America. From this position, meteorological data can be obtained over an area that includes the United States, the Caribbean area, most of South America, and a good portion of the eastern Pacific Ocean. Potential users of the data received directly from the satellite are:

- National weather centers of countries within the observational range of the satellite
- Area or regional forecast centers of the larger countries
- Military weather services of the countries
- Weather research centers.

These same stations are to be considered also as possible recipients of processed meteorological data relayed through the SMS. Consideration of another type of multiple reception station was directed by the NASA contract. This type is the Nimbus Ground Station. These are not to be confused with the Nimbus Command and Data Acquisition (CDA) Station, of which only one is planned initially. Each of the numerous stations will have a receiver and facsimile recorder, for direct readout of Nimbus cloud pictures of surrounding areas on one, two or three successive orbits. The objective here, for the SMS, is to consider the possibility of providing, for processed meteorological data, a satellite relay system which is receivable on this planned equipment with minimum additions and modifications. Important boundary conditions are thus introduced for the selection or design of the spacecraft relay equipment. NASA-Goddard personnel associated with the SMS project have indicated that reception of raw meteorological data sensed by the SMS isnot an objective for these Nimbus stations.

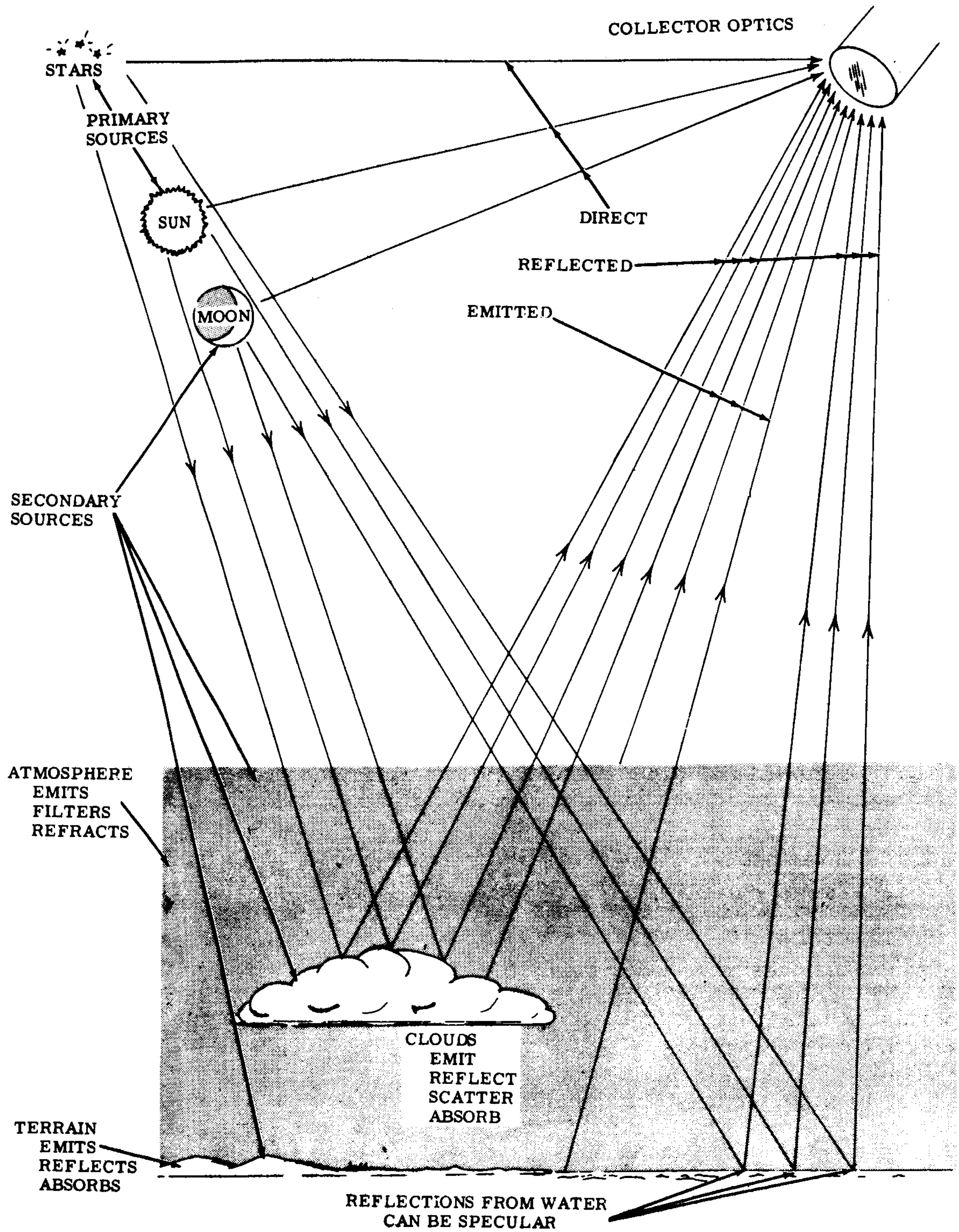

Figure 1-2. SMS System - Inputs to Sensor Channels

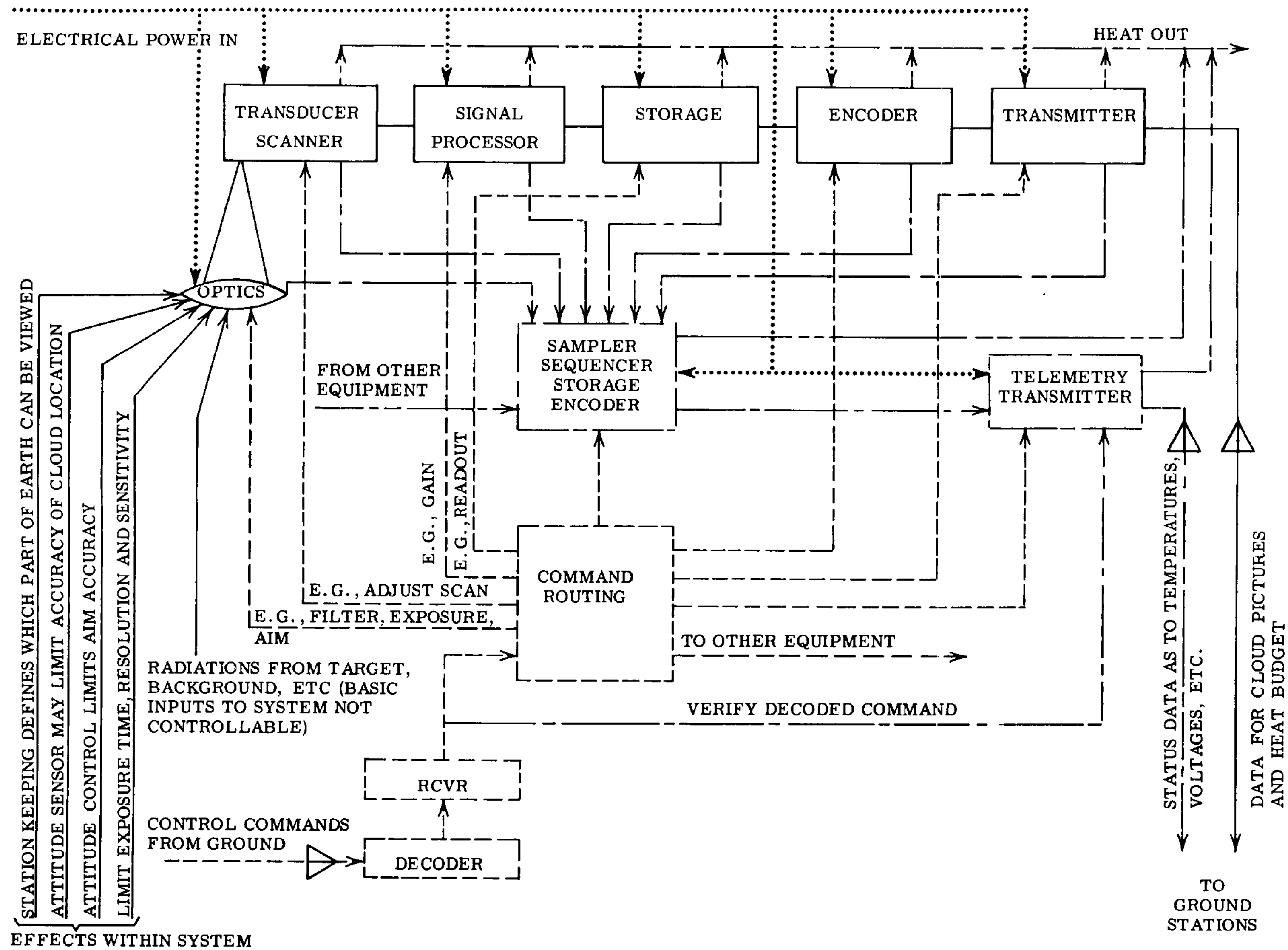

Figure 1-3. SMS System - Functions and Relations for One Generalized Sensor Channel in Spacecraft

The remainder of this section (1) is devoted to a review of objectives and functions of the system as a whole and its principal subsystems, and provides a perspective for comparing the three illustrative satellite configurations, and certain alternatives, which are presented in subsequent sections of this volume.

B. OBJECTIVES FOR METEOROLOGICAL DATA OUTPUTS

1. General

The types of meteorological data to be delivered by the SMS system were indicated above. The quality of the data that can be achieved by various SMS systems is the value scale against which the possible systems are compared. The principal quality item is resolution. Also important are dynamic range (gray scale) and sensitivity for operation at low energy levels. Location accuracy for cloud features is another important quality item.

Although systems and subsystems are to be compared over a spread of performance as to data quality, it is necessary to compare this spread with target values which reflect the needs of the users of the data. However, estimates which are useful for indicating objectives for data quality, may be obtained from experience in utilization of Tiros data. These are given below.

In this connection, the primary interest is in quality of the data as delivered to users. This involves not only the sensors but also the rest of the channel through the transmitter, receiver, and recorder, as indicated in Figure 1-3.

2. Heat Budget Measurements

The SMS will deliver data from measurements of the radiations reflected and emitted by Earth, atmosphere, and clouds. For the reflected energy, measurements of radiation intensity are needed in the wavelength band from 4 microns down through the visible region into the near ultraviolet. Emitted radiation is measured in that part of the spectrum above 4 microns. These two measurements suffice for simple heat budget calculations. For more refined calculations, additional measurements can be added, as is done in the Nimbus five channel radiometer.

The contract calls for measurements of the heat budget of the Earth (but not of individual cloud systems). It says, "The study shall provide detailed information on the variation of solar and terrestrial radiation intensity within the field of view of the spacecraft as a function of the time of day." Some meteorologists have indicated that horizontal resolution of the order of 100 to 300 mi suffices for the Earth's heat budget. It would also be desirable to measure the total radiation from the full disc of the Earth.

3. Cloud Cover Objectives

It is desirable to take cloud cover pictures with resolutions of approximately 1 mi. For interpretation of cloud features and recognition of landmarks, a gray scale of eight or more shades is required. Moreover, for day and night coverage, the equipment must be capable of functioning over the wide range of light levels that will be encountered.

4. Location Accuracy

For global or hemispheric weather analyses, location errors up to 50 mi are acceptable. For purposes of obtaining greater detail of limited areas, it is desired that the errors be considerably less.

In connection with the location problem, it may be necessary to deliver with the raw meteorological data additional data on SMS attitude angles and sensor aim angles. These data may be needed for rectification of the SMS cloud pictures or for their conversion to Mercator or other projections.

The use of landmarks was mentioned for improving location accuracy. The question arises as to how often they will be obscured by clouds. The same question arises in connection with the possible use of a map-matching technique for attitude monitoring (discussed in Section 2. A of Volume 4).

A preliminary answer is provided by study of a world chart entitled "Total Hours of Sunshine (Annual)," issued in 1955 by the Climatological Services Division of the Weather Bureau. On this chart are contours for various amounts of sunshine. Much of the data was derived from cloud summaries. The chart shows that there are many large areas which have sunshine (or the absence of clouds) for 50 to 90% of the daylight time. These areas are widely distributed within the hemisphere viewed by the SMS, and contain many landmark features and patterns. Among these areas are the Hawaiian Islands, Marquesas Islands, Tuamoto Archipelago, Society Islands, Andes Mountains between south latitudes 20 and 30°, the coast of Ecuador, the Gulf of California, much of Western United States and Mexico, much of the Gulf of Mexico-Carribean area, some of the north coast and the eastern bulge of South America, most of the coast of Argentina, the Cape Verde Islands, and the coasts of French West Aftica and French Guinea.

It is to be expected that many usable landmarks will always be visible. These landmarks, in combination with sections of horizon visible in the cloud pictures, can be utilized to improve location accuracy.

The objectives of location accuracy impose requirements for low distortion, or at least calibrated distortion, in the picture sensor and associated data links and recorders. In some Tiros pictures, variable distortion has been encountered, apparently due to noise.

One concept which appears worthy of investigation involves the addition, at the first image plane, of a grid of relative latitude and longitude, or something equivalent. This would facilitate direct reading of locations, and the reference grid would disclose any distortions introduced further along in the channel. Possibilities can be seen for modifying the scan in the recorder, or in later data processing, to convert the picture to an equal area or some other projection.

5. Relayed Meteorological Data

The primary objective for the processed meteorological data to be relayed through the SMS is simply that it not be degraded as it passes through the communication channel.

C. OBJECTIVES FOR SPACECRAFT SUBSYSTEMS

The principal subsystems required in the SMS are indicated schematically in Figure 1-4, together with a few of their interactions.

The objectives for the subsystems required in the meteorological data channels have just been reviewed. These subsystems are the meteorological sensors, the sensor data transmitter, and the relay receiver and transmitter. Sensor performance and data quality can be limited by the attitude control system. In particular, for low levels of illumination, long exposure times are needed, but they cannot be so long that the effective resolution is reduced by blurring associated with the attitude rates. These rates can arise from external disturbances, from attitude correction torques, and from motions within the spacecraft. Such rates must be damped, at least during exposures.

For the purposes of users of meteorological data, the attitude angles need be controlled only to the extent necessary for aiming the fields of view of the sensors to cover desired parts of the Earth. However, these users need rather precise measurements of attitude angles and aiming angles for accurate geographical location of cloud patterns, for rectification of cloud pictures, or for conversion of the flat pictures of the globe to Mercator or other projections.

Requirements for tighter control of orientation arise in connection with the use of thrust for purposes of station keeping and for the maneuvers and velocity changes involved in injection into orbit and adjustment of the orbit. For these adjustments, a minimum weight subsystem is needed which can provide enough energy for correction of the largest likely injection errors. Beyond this is the need for an energy supply to suffice for all of the station corrections required during a year or more of satellite operating life.

It appears that a station keeping accuracy of a few degrees of latitude and longitude will be adequate for the needs of the data users. For this study ±2° of latitude and longitude was considered a satisfactory station keeping tolerance.

For purposes or orbit injection and adjustment, station keeping, and guidance during the ascent to orbit altitude, beacon functions must be provided, with equipment to be compatible with NASA tracking facilities. Equipment provided for other purposes such as telemetry, may be made to serve this purpose.

For these phases of the satellite operation, and for monitoring the status and operating conditions of the spacecraft subsystems, a telemetry system is required. The NASA contract adds the further requirement that the telemetry system shall be capable of determining launch performance and shall be compatible with present NASA telemetry and range systems.

For control of SMS orbit injection, of orbit correction, and of observation and reporting functions, a command link is included. Provisions are required for verifying the command as received and decoded prior to execution.

A power supply system is included to meet the needs of the various subsystems. Study of several types of power systems was directed by the contract. Solar cells were chosen to supply the electrical energy for all of the illustrative satellite configurations presented in this volume.

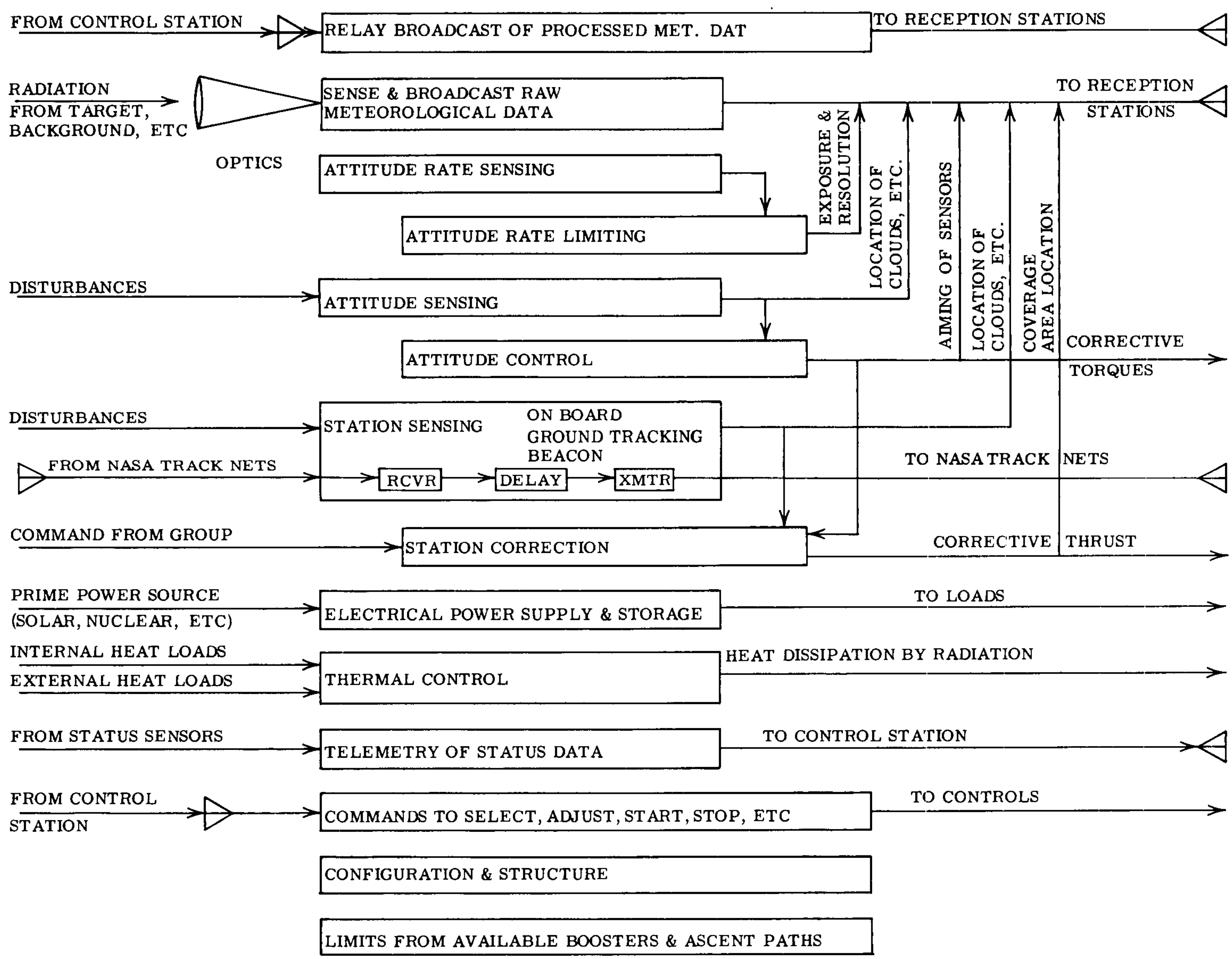

Figure 1-4. SMS System - Schematic of Principal Spacecraft Subsystem Functions

The NASA contract required a preliminary spacecraft configuration to reflect the results of the parametric studies of subsystems and their integration. This volume presents three such configurations, with additional discussions of three stabilization alternatives.

In order to increase the reliable life of the spacecraft and its subsystems, a thermal control system is utilized to hold temperatures within narrow limits in the presence of solar heating and internal heat loads.

The prime concern in Republic's studies was to maintain the SMS in a synchronous orbit. However, in accordance with the NASA contract, some consideration was given to orbits inclined 28 to 30°. These orbits offer three features which may be advantageous. First, weight is saved in the "apogee kick" motor, since a lower total impulse is required to circularize the inclined orbit. Second, control system weight and complexity are reduced. Third, the daily north-south figure-8 excursions of the satellite subpoint extend the observational coverage into polar areas, although with a corresponding reduction of continuous coverage area.

For this study, it was required that a standard national boost vehicle be used without modification for boosting the spacecraft to orbit altitude. The 100, 500, and 1000 lb weight goals chosen for the three configurations presented in this volume were chosen to fit three boosters expected to be available; these are the Thor-Delta, Atlas-Agena, and Atlas-Centaur, respectively.

D. GROUND FUNCTIONS OUTSIDE THE SMS SYSTEM

Outside of the SMS system, but important for consideration in connection with it, are the following functions:

- Boosting the spacecraft to orbit altitude.
- Injecting it into the desired orbit at approximately the desired station.
- Tracking, computation, and guidance for these functions.
- Tracking and computation for station keeping.
- Meteorological analysis and interpretation, including preparation of "processed meteorological data" for relay broadcast through the SMS.
- Reception, recording, and utilization of "processed meteorological data" relayed through the SMS, to 50 to 150 Nimbus APT-TVCS ground stations.

With regard to these functions, the scope of this study is limited to input, output, and interface problems as they affect the SMS system proper.

1. Ascent into Orbit

It is specified in the NASA contract that the SMS spacecraft is to be designed to utilize a standard national boost vehicle for its ascent into orbit, without requiring any changes in the guidance or other features of the vehicle.

It is also stated in the contract that the spacecraft telemetry system is to be capable of determining launch performance and is to be compatible with present NASA telemetry and range systems. Hence the boost phase does impose certain requirements on the spacecraft and its equipment but does not involve functions for the SMS ground system. During much of this phase, the spacecraft will be outside of the line of sight limits for ground stations located in the ultimate SMS coverage area centered at latitude 0, longitude 90°W. Present and planned NASA tracking and guidance nets will cover the flight paths contemplated.

2. Injection into Orbit

During this phase, the spacecraft may be in view of the SMS control station. If so, telemetry reception and recording could be started at the station. It might be possible to command ignition of the "apogee kick" motor from the control station, but this would involve early, short-delay delivery to the control station of data from the tracking and computation nets. Also, a transfer of control would be required. For these reasons, it appears that control of injection is a function which should remain outside of the SMS system.

The contract specified "a compatible command system for the spacecraft." It is clear that the command receiver in the spacecraft and the command transmitters, both at the SMS control station and at the injection control station, must all be compatible. Also apparent is the need, in the spacecraft, for command reception antennas with essentially omnidirectional coverage before final stabilization is achieved.

3. Tracking and Computation

Present and planned NASA stations, nets, and other facilities are adequate for the needs of ascent and injection of the SMS, as well as for many other types of satellites. Some of their capabilities have already been demonstrated. Present indications are that present and planned NASA facilities are adequate for tracking the SMS position, altitude, and velocity after injection into orbit, and for computing the velocity increments necessary to bring it to its nominal station initially, as well as those needed later for station keeping corrections. Communications to deliver these data to the SMS control station will be required, as indicated in Figure 1-1.

There appears to be no need for duplicating any of these tracking and computation facilities within the SMS system. However, after the SMS arrives on station, data for direct monitoring of station keeping may be obtainable from the aiming controls associated with the narrow beam antenna at the control station for receipt of the raw meteorological data.

4. Meteorological Functions

The primary functions of the SMS system are to acquire and deliver raw meteorological data and to accept and to broadcast "processed meteorological data." The system is to be designed as a tool for meteorologists. In their hands, outside of the SMS system, remain the functions of analyzing and interpreting the observational data; of disseminating results; and of preparing the "processed meteorological data" for SMS relay broadcasts.

E. OBJECTIVES AND FUNCTIONS OF GROUND STATIONS

1. Types of SMS Ground Stations

The NASA contract specifies two types of SMS ground stations:

(1) Control and Data Acquisition (CDA) stations.

(2) Multiple Data Acquisition (MDA) stations.

It is intended to locate one control station in North America. Its functions will be generally similar to those of Tiros and Nimbus CDA stations, and will include monitoring of spacecraft operating conditions through a telemetry link, control of these conditions and of meteorological observations through a command link, and recording of meteorological data sensed in the satellite. Unlike the Tiros and Nimbus CDA stations, the SMS station can maintain continuous contact with the satellite. Figure 1-5 is a functional schematic of an SMS control station.

A substantial number of MDA stations will be located within the near-hemispheric line of sight communications limits of the SMS. The functions of the MDA stations are limited to reception and recording of meteorological data of two types. The first type is the raw data sensed in the satellite, the same data as that recorded at the control station. The second type is the "processed meteorological data" delivered from outside of the SMS system to the control station, transmitted from there to the SMS, then rebroadcast. The second type is a control station function additional to those cited above. For both types of data, the objective is to deliver hemispheric and global weather data to weather centrals and other meteorological users having needs for them. For these stations, there are obvious possibilities for utilizing the same receivers and recorders for both types of data. A functional schematic for an MDA station is presented in Figure 1-6.

For this study, NASA directed that consideration be given to 1) use in MDA stations of receiving and recording equipment being developed for the Nimbus APT-TVCS Ground Stations, and 2) relay through the SMS of "processed meteorological data" into this equipment as built, located, and installed for the Nimbus program, with minimum modifications or additions or interference with reception from Nimbus.

As to the first consideration, it was found not feasible to use the equipment for receipt of raw meteorological data from the SMS, within the objectives set for area coverage, resolution, and frequency of reporting. The second consideration is feasible with certain limitations. For this case, the Nimbus stations can be regarded as special types of MDA stations outside of the SMS system. Clearly, use of these stations imposes restrictions on the selection or design of relay equipment in the SMS and on the types, forms, amounts, and speeds of meteorological data which can be relayed.

2. Types of Data For Relay to Nimbus Stations

The relayed data received by the Nimbus stations will represent some compromise among four factors.

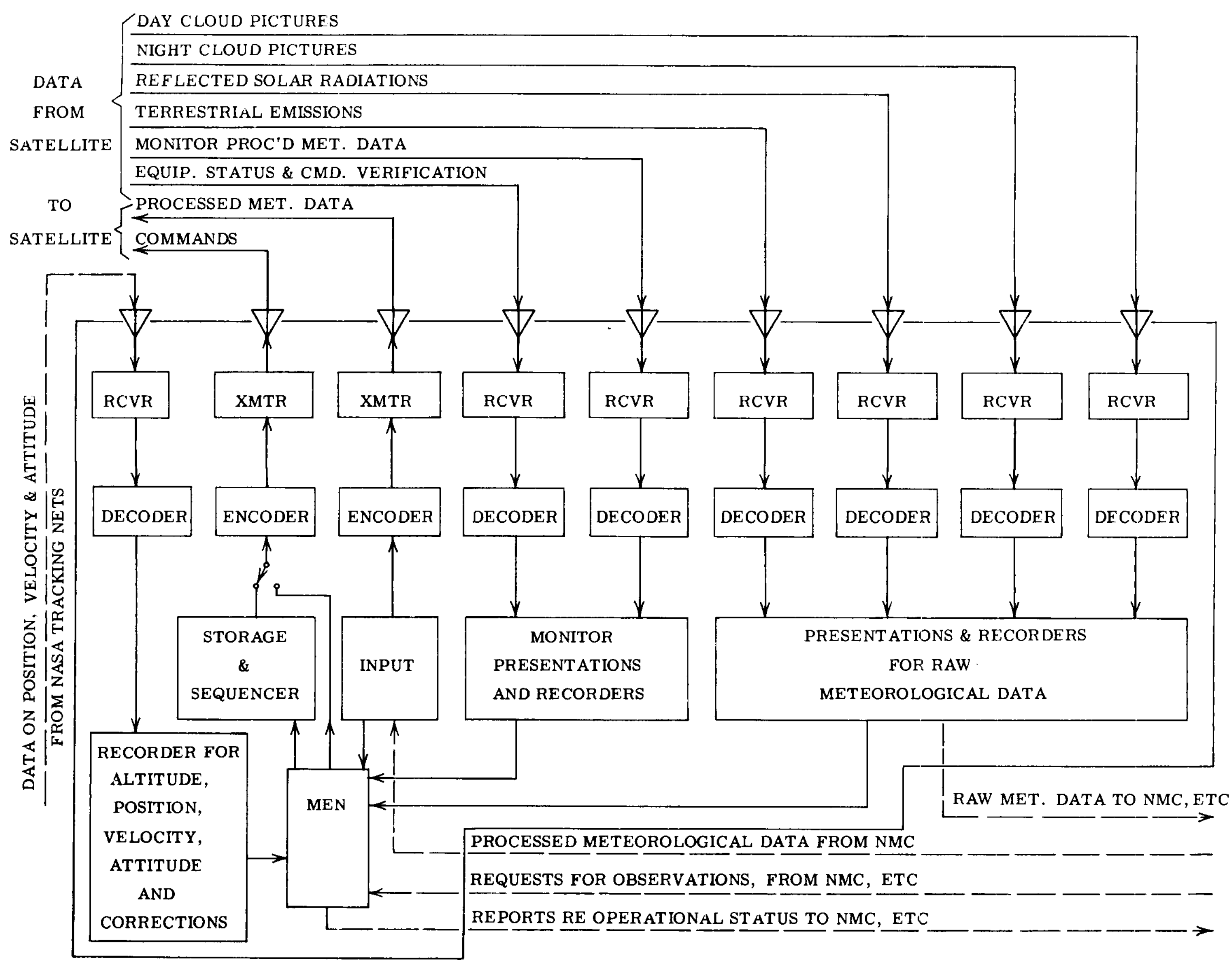

Figure 1-5. SMS System - Control Station Schematic

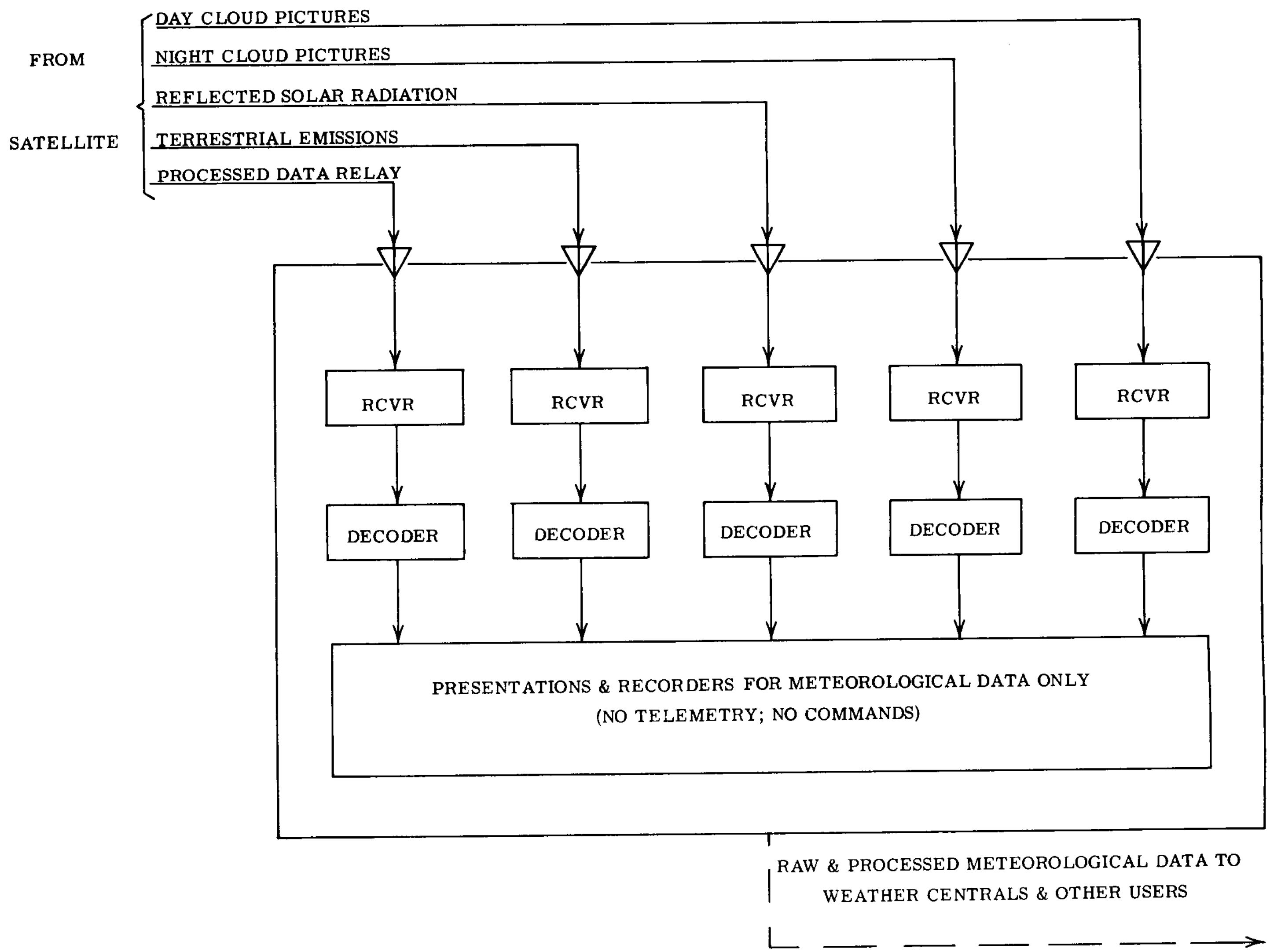

Figure 1-6. SMS System - Schematic for Multiple Data Reception Stations

(1) What is needed or wanted by users on the receiving end.

(2) What can be processed and supplied by the originating weather service.

(3) What can be received and recorded by Nimbus station equipment of fixed design and characteristics, with acceptable modifications.

(4) What can be relayed through the SMS, in the time available, by transponder equipment already there for other functions and/or which can be added.

No substantial problems are apparent in providing, at the control station, input, transmitting, and monitoring equipment suitable for data forms defined by these factors.

Investigation of meteorological requirements was explicity excluded from this study by the NASA contract. However, it is highly likely that what is most wanted by users of cloud pictures from Nimbus APT-TVCS Ground Stations will be those types and forms of meteorological data which will best facilitate and enhance utilization of these pictures. Of available materials, the types most likely to be wanted are:

(1) Conventional meteorological data or analyses for the area covered by the Nimbus pictures acquired by each station.

(2) Meteorological data or analyses for large areas additional to those covered by the pictures.

It is to be noted that these Nimbus stations will be distributed widely within the near-hemispheric coverage area of one SMS. As a minimum, it is to be expected that they will be widely spread in the continental United States. However, each will receive Nimbus pictures for only a fraction of this area. If the desires of each station are primarily for relay data to match its Nimbus coverage, then each will want data for an area different from the others. Hence, it appears that the first item to consider for relay transmission to Nimbus stations is the hemispheric weather analysis charts, centered on the United States, which are now routinely produced by the Weather Bureau and distributed by standard facsimile net to many military and civilian centers. Questions are apparent as to whether relay through the SMS will save enough time or money or reach enough additional recipients to be justified. It is to be recognized that aspects of the operational SMS system may be different from those of the R & D phase.

Another form of data which may be considered for relay to Nimbus stations is that of cloud pictures for areas beyond their own coverage. These could come from Nimbus and/or SMS observations. At best, they would depict cloud cover one to several hours earlier than current Nimbus pictures. Most likely, they would be relayed after rectification, analysis, and annotation. The recorders at the Nimbus stations are suitable for recording of photo-facsimile. Some quality degradation may be acceptable.

Processed meteorological data in the form of letters and numerals, such as teletype messages, could be transmitted by either type of facsimile relay channel, as well as by simpler channels.

3. Meteorological Data for SMS Stations

The primary functions of the SMS system are to sense certain types of meteorological data and to deliver these to users at many locations. One type of data can be called "raw meteorological data," to distinguish it from "processed meteorological data." Relay of the latter to many ground stations is another system function. Both the control station and the MDA stations must receive and record both types of data.

The raw data is of primary interest. Some objectives for quality of raw meteorological data to be delivered by the SMS system are reviewed in Section 1.B of this volume. Among the objectives for both CDA and MDA stations are receipt and recording of these data with minimum degradation of the quality available from the satellite-borne equipment, and with minimum requirements on the satellite for power levels and antenna size. High gain antennas and highly sensitive receivers on the ground are indicated. However, the NASA contract sets a limit on the size of the receiving antenna at the control station, as one having performance comparable to that of a single 85 ft dish, to be operable at frequencies up to 2300 MC. For the MDA stations, economy objectives are cited, implying smaller antennas. For both types of station, the antennas need only small spreads and rates of aiming angles, since the SMS is "stationary."

The raw data is to be received simultaneously, directly from the satellite, by both the CDA and MDA stations. In both cases, it is to include, on the same communication link, all location reference, and calibration data needed for meteorological utilization. In the operational system, the raw data will be delivered to meteorological users with minimum further processing and delay.

Relayed, processed, meteorological data is to be received and recorded at the control station only for purposes of monitoring and controlling the broadcast output of the satellite. The needs of the MDA stations, or rather of the data users whom they serve, may or may not be met by the relay data discussed above, which is limited by the characteristics of the Nimbus Ground Stations. It is clear that delivery of raw data requires a much less limited channel from the SMS to CDA and MDA stations. Hence, data of much better quality and higher rate could be relayed to the MDA stations by addition of an "up-link" of comparable quality between the CDA station and the satellite.

4. Control Station Functions

In common with the MDA stations, the CDA station records raw and relayed meteorological data. The raw data is recorded and displayed for purposes of "quality control" of the output of the spacecraft, as well as for utilization by meteorologists.

The control station has additional functions which are not performed at any of the other stations. These are:

(1) Receipt, input, and transmission to the SMS of processed meteorological data for relay.

(2) Receipt and recording of telemetered data on the status and operating conditions of the SMS and its equipment.

(3) Control of spacecraft conditions and operations, through a command link.

(4) Receipt and recording of data as to spacecraft position, velocity, and corrections needed.

Item (1) needs little amplification, in view of preceding discussions on the relay of processed meteorological data. Monitoring of the satellite relay and telemetry emissions may indicate needs for adjustments in the ground inputs on transmitters, for commanding changes in the satellite transponder, or for repeating parts of the message.

As to item (2) the telemetered "housekeeping data" is utilized immediately for commanding changes in operating conditions, biases, etc., for selections among redundant equipment elements, and for verification of decoded commands. Telemetry records may also be used for troubleshooting. Over longer time periods, these records may be reviewed for performance history of equipment, as a guide to development of improved equipment for later spacecraft.

For item (3), two types of control are to be distinguished. The first is control of meteorological observations, broadcasts, and relays. The second is corrective action as to satellite station, attitude, and internal conditions. Unlike earlier meteorological satellites, the SMS will be in continuous contact with the control station and, hence, will provide opportunities for transferring to the ground many of the functions which, in other satellites, must be either internally programmed or automatic.

Item (4) could be eliminated if the station correction commands are transmitted from a tracking facility outside of the SMS system. This procedure may be acceptable as an emergency measure. However, it seems preferable to retain the tracking function at the SMS control station in order to eliminate possible interference with meteorological observations and to utilize the pictures of the Earth as indicators of needs for station correction and of corrections achieved.

The preceding review of control station functions applies after the SMS has arrived on station. During the ascent and injection phases, the tracking net will exercise all control.

F. ILLUSTRATIVE SPACECRAFT CONFIGURATIONS

Following sections of this volume present spacecraft system configurations illustrative of the performance that can be expected in three different weight classes. All of these configurations were derived by synthesis of subsystems surveyed and analyzed in the other volumes of this report.

For the first class, a target weight of 100 lb was chosen as an example of a minimum weight satellite which could be put into a synchronous orbit by a Thor-Delta booster. A cylindrical configuration, spin stabilized about an axis parallel to that of the Earth's polar axis, was found to provide the lightest configuration. Spin introduced problems of exposure timing and image motion compensation. Even with rather limited capabilities for acquiring meteorological data, the weight of this satellite came out to be about 250 lb, too much for the Thor-Delta, but well below the limit for the Atlas-Agena, the next larger available booster.

The Atlas-Agena defined the target weight for the second illustrative configuration at about 500 lb. Three-axis, Earth pointing stabilization is recommended for this satellite. A substantial improvement over a minimum weight satellite is achieved in the quality and variety of meteorological data obtainable.

As an illustration of further improvements to be expected later, by use of the Atlas-Centaur booster, a third configuration was derived, aimed at a total weight in the 700 to 1000 lb range. Stabilization, configuration, and data output quality are generally similar to those of the preceding configuration, but energy storage for sensor operation through shadow periods is provided, as is some redundancy in meteorological sensors, communications, and stabilization equipment.

Solar power is chosen for all three cases. Stabilization alternatives, including gravity gradient, are reviewed.

Republic is delivering a quarter-scale model of the 500 lb spacecraft to NASA-GSFC under this contract.

SECTION 2 - MEDIUM CAPABILITY SPACECRAFT

A. MISSION OBJECTIVES

The goals set for a medium capability satellite were those that could reasonably be expected of a satellite to be placed in synchronous orbit by an Atlas-Agena Launch Vehicle with those sensors either presently available or representing a conservative advance in technology. The types of information desired from the system were:

(1) Full Earth disc daylight picture

(2) High resolution picture of sectors of the Earth in both daylight and nondaylight conditions

(3) Heat budget measurement of both emitted and reflected radiation with best resolution obtainable by uncooled sensors

(4) Ability to relay selected processed information to multiple ground stations

(5) One complete set of information every 30 minutes

Equipment for the vehicle is to be compatible with existing tracking and ground stations. An effort shall be made to use existing qualified equipment where it does not compromise the vehicle design.

B. SPACECRAFT CONFIGURATION

The configuration chosen for the spacecraft is based on a 3-axis stabilized Earth oriented satellite. Three-axis attitude control was selected because it allows the sensors and data link antenna to continuously view the Earth. It also enables effective and economical integration of the functions of initial orientation, injection error correction, station keeping, and stabilization. Satellite power is obtained from an oriented solar array which rotates about the pitch axis. Nickel cadmium batteries supply power for peak demands during normal operation and during the occult period. Power for necessary functions during the ascent to synchronous orbit is supplied by silver-zinc primary batteries.

The satellite consists of two geometrically dissimilar sections as shown in Figures 2-1 and 2-2. The upper section is a hollow cylindrical section housing the apogee motor, telemetry, power supply, attitude control, passive despin weights, and solar array supports. The equipment compartments are of regular shape and size and are of sufficient volume to house many existing components (including Nimbus half modules). The arrangement provides inspection and access to the more sensitive electronic equipment through the easily removable "shadow boxes" which are incorporated for thermal control. Other less sensitive equipment (such as gas bottles and reaction wheels) is accessible through doors in the outer skin.

The upper end of this section terminates in a structural frame that supports the despin mechanism and is also the means of attaching the satellite to the booster spin table. The lower end of the cylinder is closed by a bulkhead on which the sensor equipment is mounted.

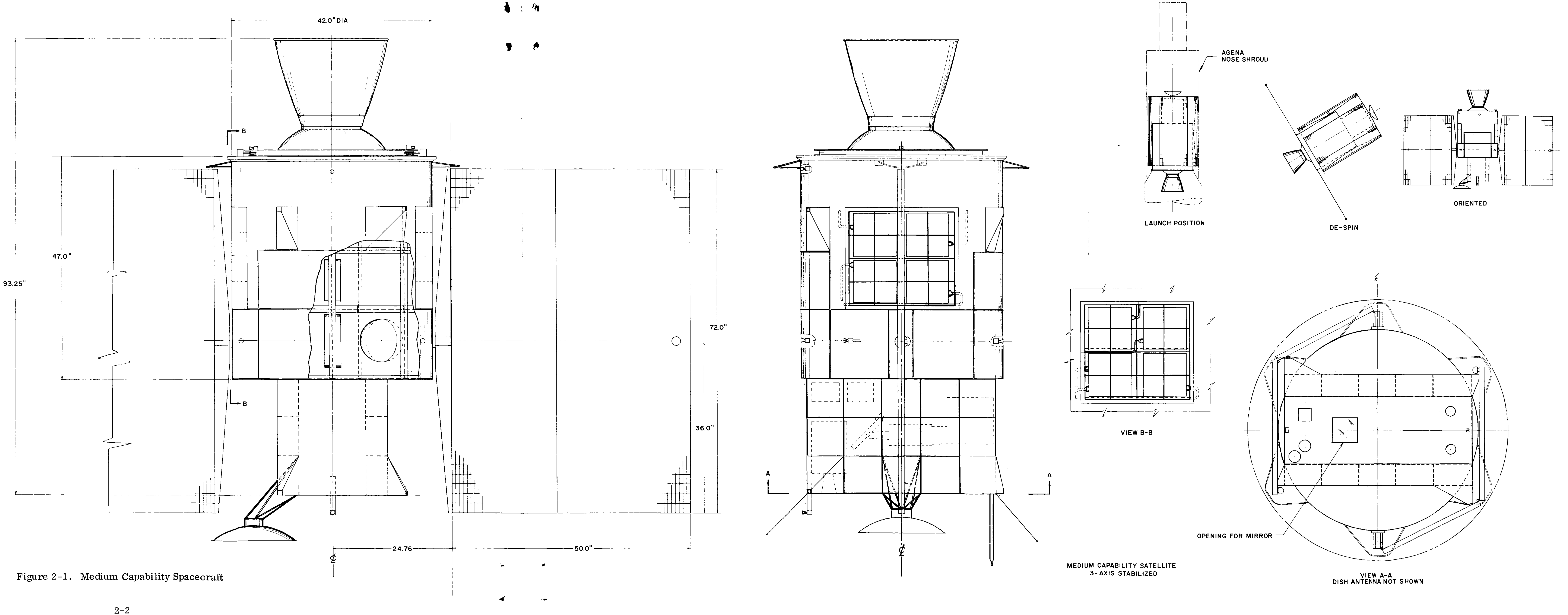

Figure 2-1. Medium Capability Spacecraft

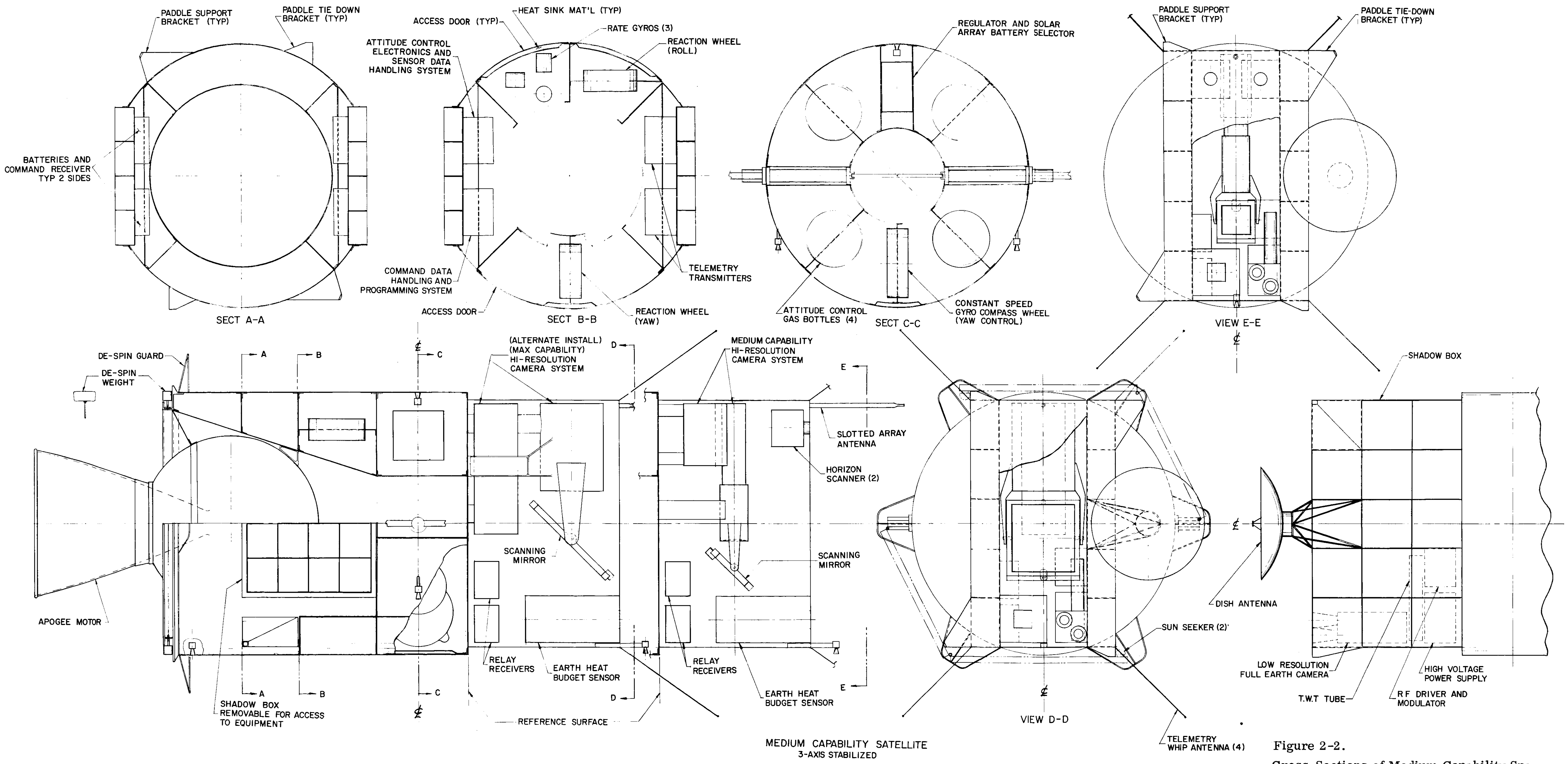

Figure 2-2.
Cross-Sections of Medium Capability Spacecraft

The sensor equipment section is roughly rectangular. It is designed to house a specific set of equipment, rather than to provide quick interchangeability for alternate sensors. Studies of sensors indicate a diversity of physical size and weight. They require a range of mounting provisions that rule out interchangeability. It is considered that the proposed arrangement permit the installation of an alternate equipment section with minimum disruption of the basic vehicle design.

The satellite has been designed to maintain proper equipment temperature by means of passive thermal control. This is made possible by using proper surface coatings, oriented panels, insulation, local heaters, and heat sinks. The considerations for thermal control have dictated the configuration in many areas. Space is available about the cg for installation of an air bearing for test of the attitude control system. Provisions have been made for approximately ±15° motion in any direction. For this test it will be necessary to remove the apogee motor and simulate its mass.

Tables 2-1 and 2-2 summarize the weight and power requirements of the satellite.

TABLE 2-1
WEIGHT SUMMARY
MEDIUM CAPABILITY SATELLITE

Item		Weight (lb)
Sensors		83.0
Communications		22.0
Data Handling		29.5
Attitude Control		103.0
Power Supply		127.7
Thermal Control		10.0
Structure		126.8
Wiring		22.0
	Total spacecraft weight	524.0
Apogee Motor		745.0
Adapter		42.0
	Total weight at launch	1311.0

TABLE 2-2
POWER SYSTEM SUMMARY
MEDIUM CAPABILITY SPACECRAFT

Primary Battery		18.2 lb
Secondary Battery		21.5 lb
Solar Array		63.0 lb (50 ft^2*)
Regulator-Selector		12.0 lb
Solar Paddle Drive		3.0 lb
	Total	177.7 lb

* Required Solar Cell Area = 43 ft^2

System Requirements	Acquisition (W-hr)	Track Average (W)	Track Peak (W)
Attitude Control	79W x 1 hr	105.0	241.0
Communications	60W x 6 hr	50.0	75.0
Data Handling	31.1W x 6 hr	33.1	33.1
Power Supply	7W x 6 hr	7.0	7.0
Sensor Equipment	10W x 6 hr	77.1	81.0
	727.6	272.2	437.1

C. MISSION PERFORMANCE

The satellite configured for the medium weight and performance range has explicit capabilities outlined further in this section. For the most part, it has equipment that is presently available or under immediate development. Present technology is considered capable of developing the required sensors. Life expectancy is one year in orbit. Reliability is expected to be good but could be improved by duplicating certain critical systems such as attitude control and sensor electronics.

The electrical power supply is not marginal - approximately 18% excess has been provided for damage and degradation, based on minimum solar power conditions. (A solar cell efficiency of 6% was assumed.) Gas for the attitude control system has been figured on conservative assumptions, and then increased by 50%. The estimated weight for the system allows approximately 100 lb for contingency or growth, based on the weight that the Atlas-Agena can put in synchronous orbit. Further growth is possible by developing techniques described in this volume.

The performance of the satellite is outlined here, with detailed descriptions of the subsystems given in subsection D. The sensors will provide:

(1) Full Earth disc picture using a vidicon camera
Resolution at the nadir* - 7 statute mi per TV line
Dynamic range - daylight conditions
Cycle time - one picture every 30 minutes
Shades of gray - 8

(2) High resolution picture using an image orthicon camera and two axis scanning mirror
Resolution at nadir - 1.3 mi per TV line
Dynamic range - daylight to 1/2 Moon
Coverage per frame - 1250 x 1250 statute mi
Frames per cycle - up to 25
Shades of gray - 8

(3) Heat budget measurements using an infrared detecting device
Full Earth coverage with 200 statute mi resolution
Emitted radiation of 4 to 40 microns in range of 175 to 325°K
Reflected radiation in the 0.2 to 4 micron range

(4) Communications relay
Will receive and retransmit data up to 100 KC base bandwidth on S-band (1700 to 2300 MC). The sensor data transmitter is used for this purpose (using time sharing or multi-frequency carrier operation).

The 3-axis stabilized control system will maintain the pointing accuracy to 1°. The pointing accuracy is limited by available horizon sensors because the basic system, otherwise, is capable of a 0.1° accuracy. In normal operation, roll and pitch errors are determined by horizon sensors, and corrective control torques are produced by variable speed reaction wheels. In the yaw axis (Earth-pointing axis) a constant speed wheel performs the functions of sensing yaw errors and producing corrective torques. Cold gas jets are used for unloading the pitch and roll reaction wheels and station keeping. The spacecraft is maintained on station within ±2° in latitude and longitude.

The overall system is shown in the block diagram, Figure 2-3. The command receiver and telemetry transmitter operate at VHF, 148 MC and 136 MC, respectively. Sensor data and the communications relay operate at S-band.

For the sensor data transmitter the base bandwidth is 100 KC. The parabolic reflector antenna on the spacecraft is 21 in. in diameter and the ground antenna diameter is 85 ft. The system provides an output S/N of 46 db and a safety margin of 19 db. The communications relay uses the same transmitter and antenna in the spacecraft but assumes a 30 ft diameter ground antenna. This gives a 31 db S/N ratio and a 7.5 db safety margin.

The telemetry system uses PCM-FM with an information capacity of 200 bits/sec. The carrier to noise ratio, C/N, is 20 db.

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* As used herein, the term nadir refers to the point on the Earth directly below the satellite.

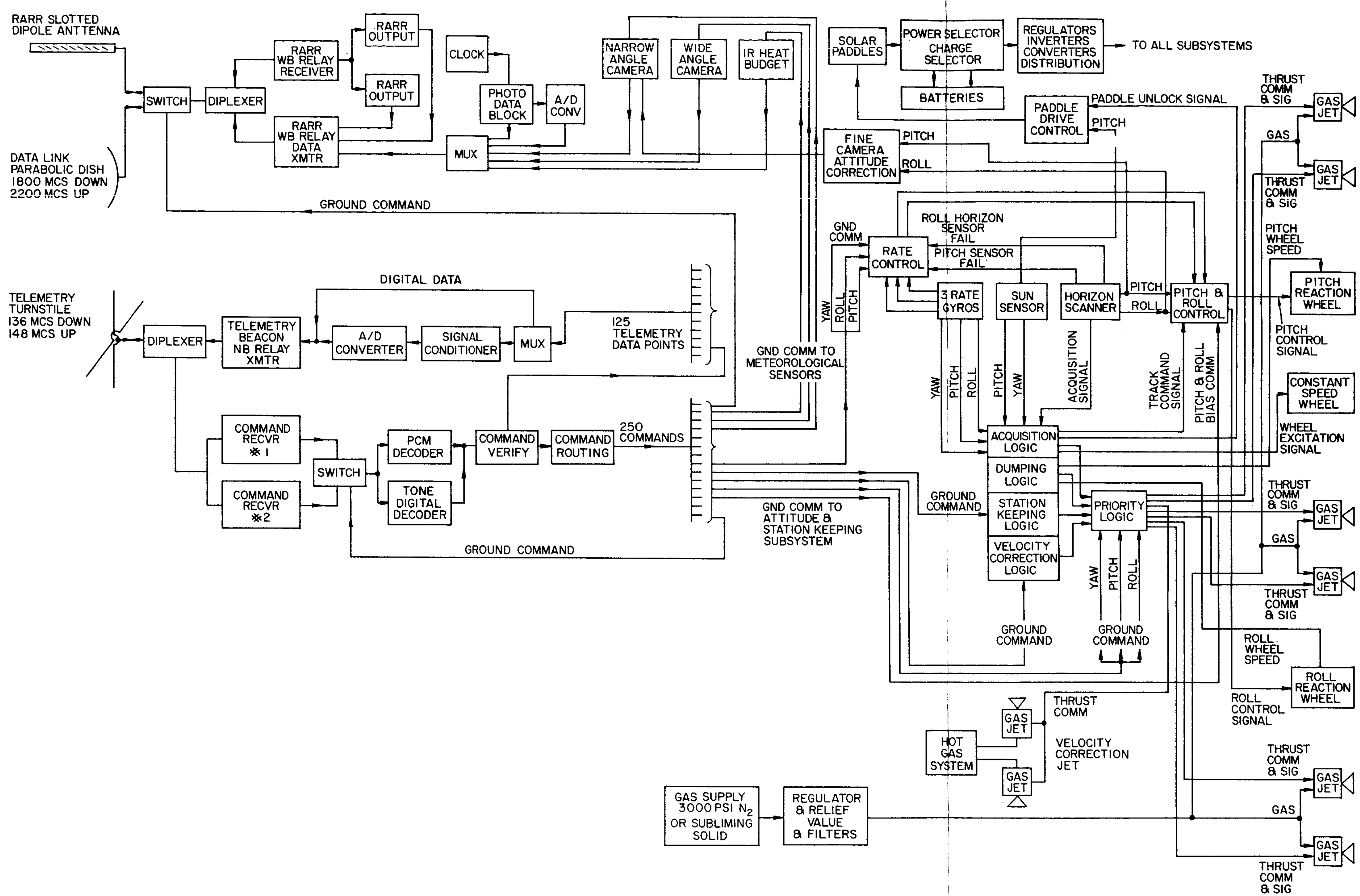


Figure 2-3

Medium Capability Spacecraft Subsystems

D. SUBSYSTEMS DESCRIPTIONS

1. Meteorological Sensors

The primary function of the SMS meteorological sensors is to provide measurements of the Earth's heat budget and surveillance of the Earth's cloud cover. These functions will be provided by three representative sensor systems which have been configured to accomplish the meteorological tasks. The sensor systems are the following:

(1) Wide coverage daytime cloud cover sensor system

(2) Narrow coverage day-night cloud cover sensor system

(3) Heat budget measurement sensor system

A description of each sensor system, its performance capability, size, weight, power and other pertinent and salient characteristics are described in the following paragraphs.

a. Wide Coverage Cloud Cover Sensor System

The wide coverage cloud cover sensor system will provide a field of view encompassing the full Earth disc and will enable the surveillance of cloud formations under essentially daytime conditions at a nadir resolution of 7 mi. This system uses a vidicon imaging tube as the sensing element and a fixed, short focal length lens. A diagrammatic representation of this system is illustrated by Figure 2-4.

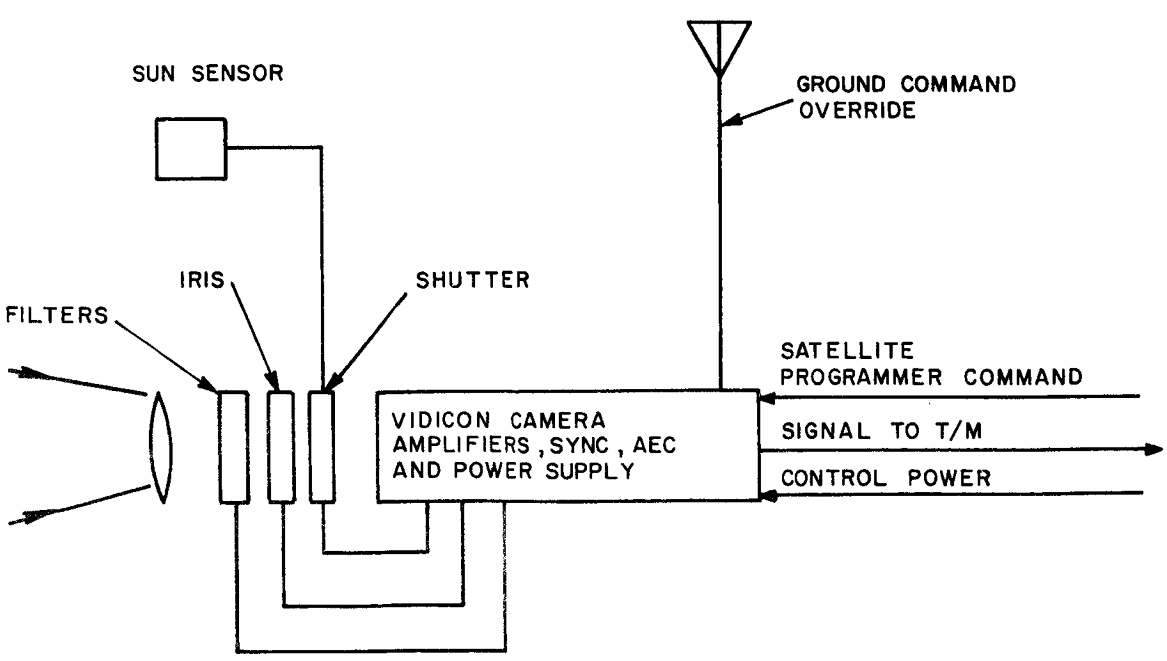


Figure 2-4. Wide Coverage Cloud Cover Sensor

The performance characteristics of this system are summarized in Table 2-3.

TABLE 2-3
WIDE COVERAGE CLOUD COVER SENSOR, PERFORMANCE

Item	Characteristic
Sensor	1 in. vidicon
Spectral Response	Visible region
Area Coverage	Full Earth disc
Resolution	7 statute mi at nadir per TV line
Number of TV Lines	800 minimum
Number of Gray Scales	8 steps of $\sqrt{2}$ difference
Operational Mode	Daytime surveillance
Operational Cycle	1 picture every 30 minutes
Dynamic Range	35 to 1 minimum/frame
Automatic Exposure Control	Filter wheel and iris
Automatic Sun Protection	Capping shutter
Satellite Stability Requirement	0.003°/sec

The sensor system provided is an integrated vidicon camera system which is made up of the following major components:

Camera Head Subassembly - This subassembly includes the vidicon camera tube, deflection components and preamplifier.

Control Chassis Subassembly - This subassembly contains the circuits and power supply necessary for the operation of the vidicon, and supplies the video signals to the sensor data handling system.

Automatic Exposure Control Subassembly - This subassembly controls the operation of a neutral density filter wheel and lens iris diaphragm for proper illumination levels at the vidicon photocathode.

Optics - An f/1.5, 1.25 in. focal length refractive lens is used.

Sun Protection Shutter - A focal plane shutter is provided to protect the vidicon camera tube when there is danger of direct exposure to the Sun's image. Shutter operation is initiated by a signal from a Sun sensor.

The volumetric, weight and power requirements have been estimated as follows:

Volume	-	375 cu in.
Weight	-	10 lb
Power, peak	-	15 W

b. Narrow Coverage Cloud Cover Sensor System

The narrow coverage cloud cover sensor system will provide an instantaneous field of view which encompasses more than 1,500,000 square statute miles of surface in a single picture and will enable both day and night-time cloud cover surveillance. The sensing element of this system is a miniaturized image orthicon tube. A scanning or aiming mirror directs the narrow coverage optics to the desired area of interest. A reflective optical system produces a resolving capability of 1.3 mi at the nadir. The major constituents of this sensor system are depicted by Figure 2-5.

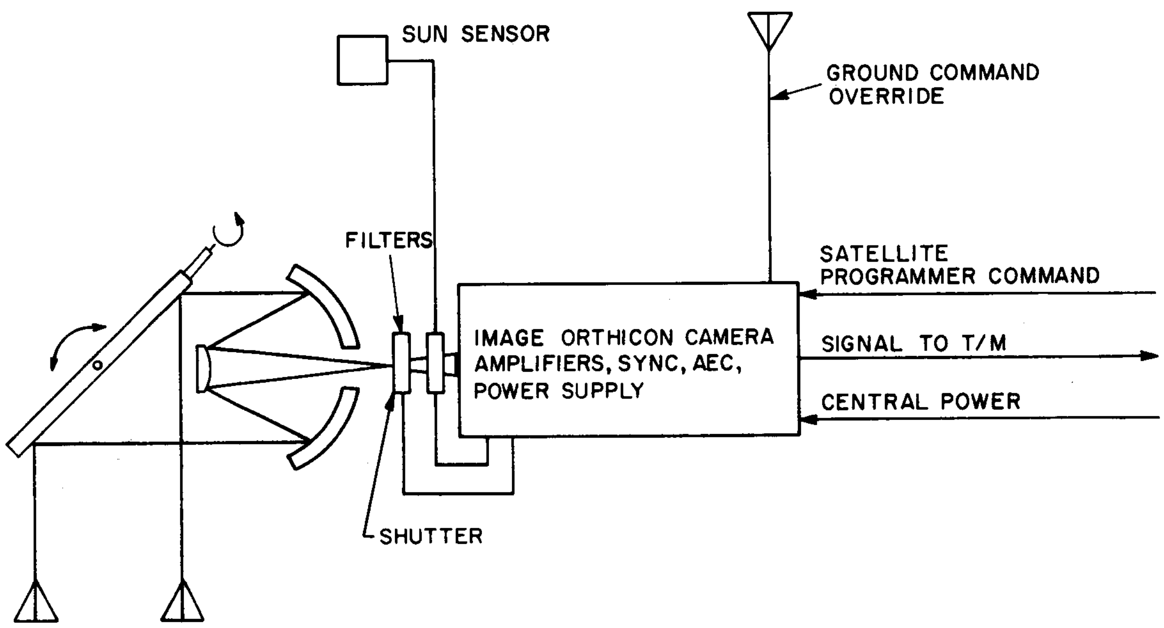


Figure 2-5. Narrow Angle Camera

The performance characteristics of this system are summarized in Table 2-4.

TABLE 2-4
NARROW COVERAGE CLOUD COVER SENSOR

Item	Characteristic
Sensor	Miniaturized image orthicon
Spectral Response	Visible region
Area Coverage	1250 x 1250 statute mi
Resolution	1.3 statute mi at nadir per TV line
Number of TV Lines	1000 lines
Number of Gray Scales	8 steps of $\sqrt{2}$ difference
Operational Mode	day and nighttime surveillance
Operational Cycle	Up to 25 pictures every 30 min
Dynamic Range	Daylight to 1/2 Moon
Automatic Exposure Control	Filter wheel and internal tube functions
Automatic Sun Protection	Capping shutter
Satellite Stability Requirement	0.003°/sec

The optics consist of the following component assemblies:

Lens - An f/4.0, 14 in. focal length reflective system is used

Scanning Mirror - Gimbaled, x-y, first surface mirror, positioning range, ±4° in each axis.

Sun Protection Shutter - A focal plane shutter is provided to protect the image orthicon tube when there is danger of direct exposure to the Sun's image. Shutter operation is initiated by a signal from a Sun sensor.

The image orthicon camera system is made up of the following major components:

Camera Head Subassembly - This subassembly consists of the image orthicon tube, deflection components and preamplifier.

Control Chassis Subassembly - This subassembly contains the circuits and power supply necessary for the operation of the image orthicon and supplies the video signals to the sensor data handling system.

Automatic Exposure Control Subassembly - This subassembly controls the operation of a neutral density filter wheel and the image tube internal functions for proper illumination levels at the photocathode of the image orthicon.

The volumetric, weight and power requirements for the narrow coverage sensor system are estimated to be as follows:

Volume	-	890 cu in.
Weight	-	43 lb
Power, peak	-	46 W

c. Heat Budget Measurement Sensor System

The meteorological sensor system used for the measurement of the Earth's heat budget is shown in Figure 2-6. This system uses thermistor bolometers to measure both emitted and reflected radiation. An optical system together with a scanning mirror is used to scan the Earth's surface. The Earth scan encompasses a 200 mi wide area that is repeated until the entire Earth disc has been covered.

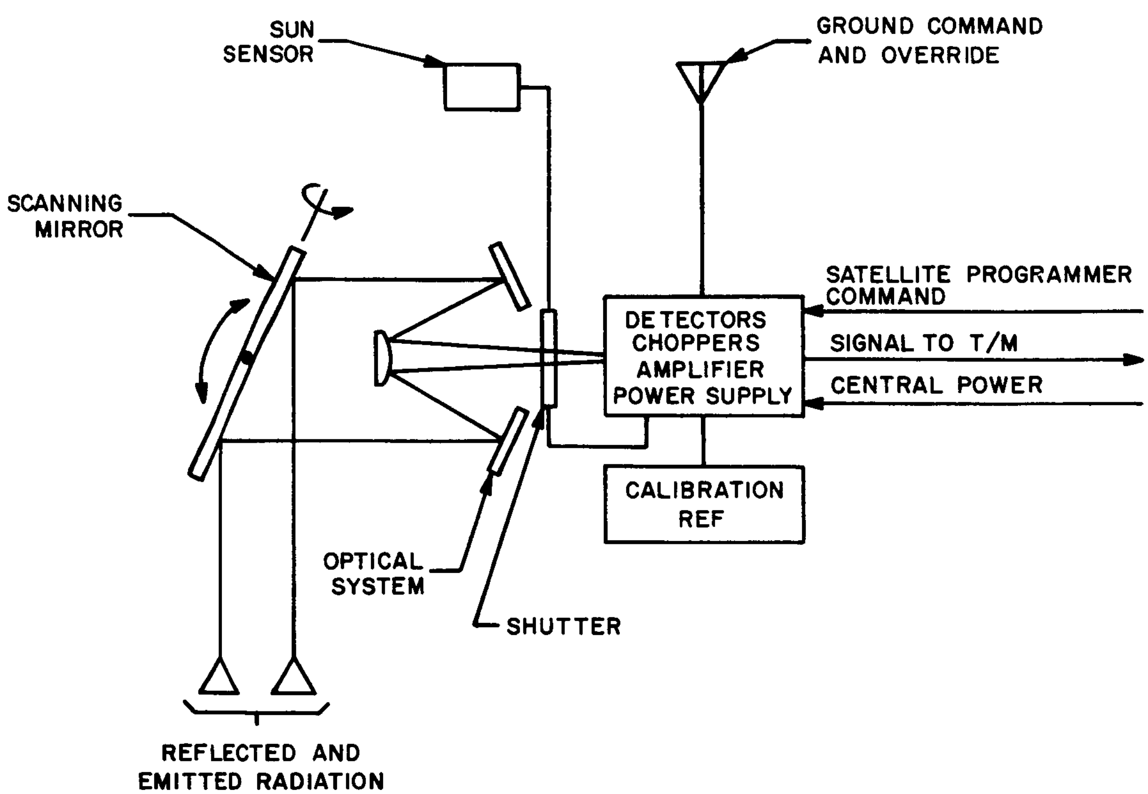


Figure 2-6. Heat Budget Sensor

The salient performance characteristics are listed in Table 2-5.

TABLE 2-5
HEAT BUDGET MEASUREMENT SENSOR PERFORMANCE

Item	Characteristic
Sensor	Thermistor bolometers
Spectral Response	Visible and infrared
Area Coverage	200 x 8000 statute mi per scan
Temperature Resolution	1.0° minimum
Number of Scans for Full Earth	40 minimum
Operational Mode	Day and night
Operational Cycle	1 full Earth coverage every 30 min
Operational Time/Cycle	1 minute max
Automatic Sun Protection	Capping shutter
Satellite Stability Requirement	0.003°/sec
Dynamic Range	175°K to 325°K
Absolute Temperature Reference	Internal black body 2°K

The heat budget sensor comprises the following major components:

Optical System - An f/2.0, 4 in. focal length reflective optics system is used.

Scanning Mirror - Gimbaled x-y, first surface mirror.

Sun Protection Shutter - Capping shutter is provided to prevent the Sun's image from impinging onto the system detectors. Shutter operation is initiated by a signal from a Sun sensor.

Sensor Head Subassembly - This subassembly consists of the two thermistor bolometer detectors, beam divider and chopper drive unit.

Control Chassis Subassembly - This subassembly contains the circuits, amplifiers and power supply necessary to process the radiation energy data into suitable form for input into the sensor data handling system.

Calibration Reference Subassembly - This subassembly provides the internal black body reference for system calibration.

The volumetric, weight and power requirements for this system have been estimated as follows:

Volume	-	825 cu in.
Weight	-	30 lb
Power, peak	-	60 W

2. Communications

The communications system aboard the spacecraft will permit transmission of sensor data to the ground, transmission of commands to the SMS, transmission of telemetry data to the ground, and retransmission of relay data and range rate data from the ground to the satellite and back to Earth. The subsystem will provide the following facilities:

(1) VHF command data receiver

(2) VHF telemetry data transmitter

(3) S-band meteorological sensor data transmitter

(4) S-band relay data receiver for reception and retransmission through a separate RF coherent translater driver channel using the sensor data wideband output TWT stage.

(5) S-band range and range rate tracking data retransmission to the ground using the same equipment as in (4) above.

(6) A multi-element S-band antenna system using a nondirectional slotted dipole array during the ascent phase and a directional parabolic antenna during the on-station phase.

(7) A common VHF omnidirectional antenna array for use by the command receiver and telemetry transmitter.

a. S-band System

1) System Performance. The S-band equipment proposed herein for the medium capability SMS will provide the following performance characteristics while the vehicle is in orbit and on-station:

Characteristic	Ground Link Sensor Data	Ground Link Relay Data
Base bandwidth	100 KC	100 KC
Output S/N	46 db	22 db
Ground Antenna Size	85 ft	30 ft
Safety Margin	19 db	7.5 db

In addition to the above capability, the system will provide a coherent response to the standard range and range rate interrogation during the ascent phase.

The system will allow for simultaneous transmission of both sensor data and relay data, but the power figures used for determination of the overall power requirements are based upon a 50% duty cycle. A block diagram for the entire system is shown in Figure 2-7.

2) Sensor Data Modulator and Driver. The FM generator will be a direct frequency modulator type discussed in Section 4.a of Volume 5. A frequency multiplier consisting of two low power cascaded varactor quadrupler stages, developing approximately 8 milliwatts at 1800 MC, will be used. This unit will feed the TWT through a ferrite isolator and power combining unit.

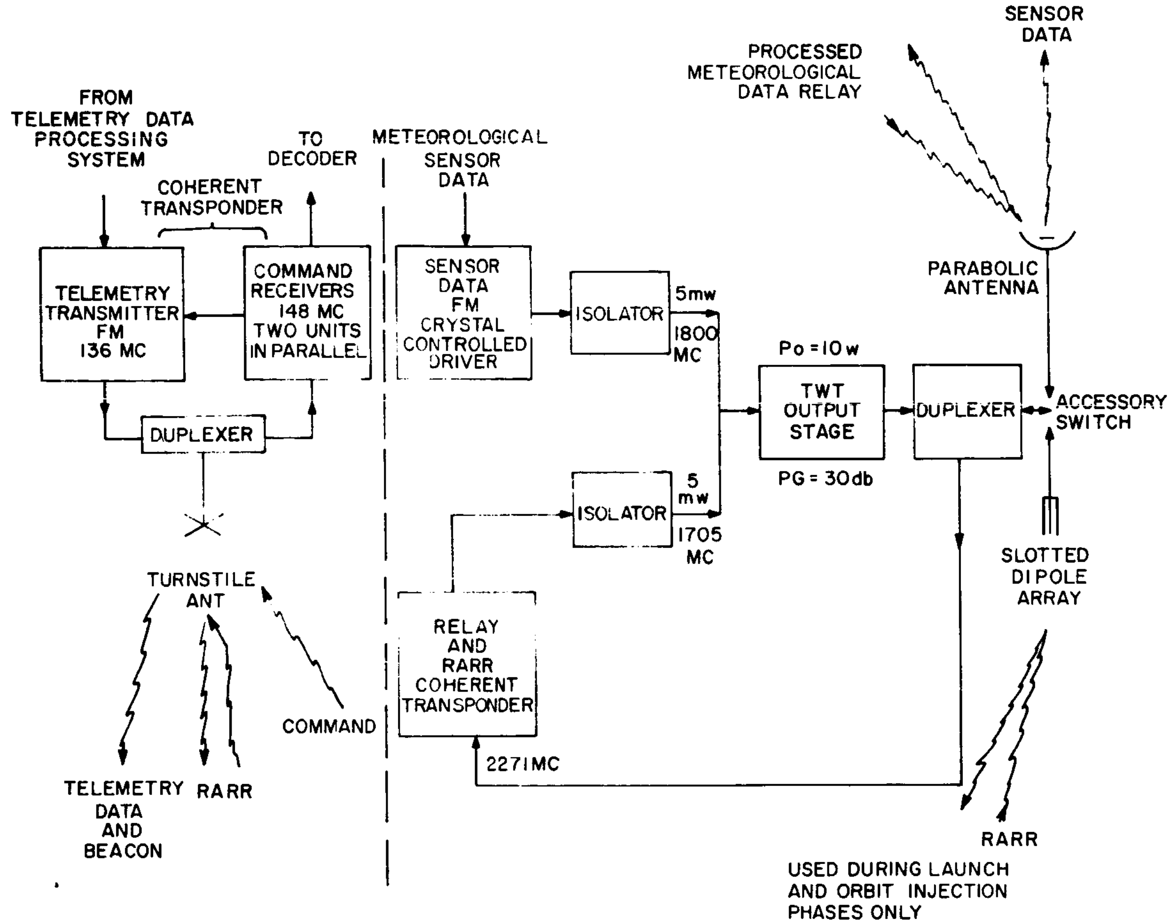


Figure 2-7. Communication System

The significant characteristics of this equipment are listed below:

Driver Output Frequency	1800 MC
Driver Power Output	8 MW
Driver Size	5 in. x 5 in. x 3 in. high
Weight	2 lb
Base Bandwidth	100 KC
Modulation Index	16.3

3) Relay and Range Rate Coherent Transponder. The relay and range and range rate functions will be performed by the same equipment. This equipment will consist of an S-band receiver in conjunction with a frequency translative type of coherent transponder. This unit, in turn, will drive a harmonic generator chain consisting of two low power cascaded varactor quadrupler stages (similar to that used in the sensor data chain). The unit will be all solid

state with a tunnel diode input pre-amplifier. This unit will have the following characteristics:

Receiver noise figure	5 db
Receiver input frequency	2271 MC
Modulation Index	2
Maximum base bandwidth	100 KC
Receiver IF bandwidth	600 KC
Relay driver power output	8 MW
Relay driver output frequency	1705 MC
Receiver volume	100 cu in.
Receiver weight	2 lb
DC power input	5 W

4) Final Power Output Stage. The common output stage for the sensor data and relay transmitter drivers will consist of a traveling wave tube (TWT) capable of developing 10 W saturated output. Two ferrite load isolators and a power combining unit will be used to couple the two driver outputs into the TWT input with minimum VSWR and interaction.

The size and weight characteristics of this stage are listed below:

TWT power output	10 W
TWT power gain	30 db
Sensor data power output (1800 MC)	5 W
Relay or range and range rate power output (1705 MC)	5 W
TWT size	2 in. x 13 in.
TWT weight	1.75 lb
High voltage power supply size	8 in x 5 in. x 5 in.
High voltage power supply weight	6 lb
Load isolator and duplexer weight	2 lb
Total Stage Weight	9.75 lb

5) S-band Antenna System. The S-band antenna system consists of a duplexer, antenna selector switch, slotted dipole array, and parabolic antenna.

As stated in the introduction the slotted dipole array will be used only during the ascent phase for determining Range and Range Rate; whereas the parabolic reflector antenna will be used when the SMS is on-station.

The selection of the latter antenna is made by the microwave selector switch on command from the ground.

The parabolic antenna will use a single wide band feed covering the range from 1700 to 2200 MC. The duplexer will provide adequate attenuation of the transmitted power to the receiver input, and permit two way operation.

The two antenna system characteristics are listed below:

Type Antenna	Parabolic Reflector	Slotted Dipole Array
Size	21 in. dia	3/4 in D x 15 in L
Weight	3.5 lb	0.5 lb
Beamwidth	21°	nondirectional pancake beam
Gain	18 db	4 db max

b. VHF Subsystems

1) Telemetry Transmitter. The telemetry transmitter of the medium capability SMS will have the characteristics shown in Table 2-6. The unit will be a solid state device having an output of 2 W, and operating at 136 MC. The telemetry system will use PCM-FM with an information capacity of approximately 200 bits per sec.

The telemetry antenna will be a turnstile type having omindirectional characteristics. It will consist of four spring mounted quarter-wave whips (approximately 22 in. long) at 90° intervals around the satellite circumference. The same antenna would be used for the command receiver. A duplexer system would be used for both telemetry transmission and command reception.

TABLE 2-6
CHARACTERISTICS OF THE TELEMETRY TRANSMITTER FOR THE MEDIUM CAPABILITY SATELLITE

Transmitter	
Frequency	136 MC
Power Output	2.5 W
Weight	1.4 lb
Volume	23 cu in.
Power Input	10 W
Carrier-to-Noise Ratio (C/N)	20 db
Output S/N for deviation index of 2	27 db
Antenna	
Omnidirectional turnstile - approximately 22 in. long	4 quarter-wave whips
Weight including duplexer	0.5 lb
Frequency range	136 to 148 MC

2) Command Receiver. The command receivers of the medium capability of the SMS will be a solid state unit operating at a frequency of 148 MC. It will be an AM superheterodyne type with single conversion. The pertinent characteristics of this unit are given in Table 2-7. Two parallel units will be used for redundancy.

The antenna for the command receiver will be the same as that for telemetry, i.e., the turnstile. A duplexer will be used to permit concurrent command reception with telemetry transmission.

TABLE 2-7
CHARACTERISTICS OF COMMAND RECEIVER

Frequency	148 MC
IF bandwidth	60 KC
Image rejection	60 db
Local oscillator stability	±3.5 KC
DC power input	0.5 W
Weight	10 oz
Size	10 cu in.

c. Summary of Subsystem Characteristics

Listed in Table 2-8 are the weight, volume and power drain of the major units contained in the communications subsystem.

TABLE 2-8
COMMUNICATIONS SYSTEM POWER SUMMARY

	Quality Item	Total Weight	Total Volume	Peak DC Power Input
2-	Command Receiver	1.25	20	1.0
1-	Telemetry Transmitter	1.4	23 cu in.	10.0
1-	Relay Transponder	3	100 cu in.	5.0 W
1-	Sensor Data Driver	2	75 cu in.	5.0 W
1-	S-band TWT Output Stage	1.75	50 cu in.	
1-	TWT Power Supply	6	200 cu in.	50.0
1-	Turnstile Antenna	0.3		
1-	Slotted Dipole Array	0.5		
1-	Parabolic Antenna	3.5		
	Isolators, Duplexors, etc.	2.5		
	Total	22.2	468 cu in.	71 W

It should be noted that the medium capability configuration system weights and volumes allow for no redundancy except in the case of the command receivers.

3. Data Handling

The general area of data handling covers the arrangement of information into a format suitable for transmission and reception. Data originating at various sensors may be converted in form from one analog quantity to another more suitable one, or to a digital quantity. Because a single channel is used for many sensors, a method of channeling many sources of data to one point (multiplexing) in some predetermined sequence is included in the system. At the receiving end, a method for routing the data from a single channel to each of the required destinations (demultiplexing) in some desired sequence is needed. Other areas such as the need for reference points for synchronizing the transmission and reception of data, and for interpreting their significance come under this general heading.

a. Sensor Data

The data handling portion of the medium capability satellite is governed by the basic requirement for 7 mile resolution at the nadir for the full Earth disc picture. Using a frame which is 8000 miles by 8000 miles, there are 640,000 picture elements to be transmitted per frame. The high resolution picture requires the transmission of 390,625 picture elements; and the heat budget sensor which uses the channel only once each half-hour for a 10 to 15 sec period has a rate of 18,000 bits per sec. Because the readout period for the visual data is 10 sec, the governing data rate is 64,000 picture elements per sec. It has been determined that this will be handled in analog form. Digital data for the photointerpreter will be inserted at the bottom of each frame as discussed in Section 2. A. 4 of Volume 5.

The equipment needed to carry out the data handling function consists of a multiplexing unit which cuts in the photointerpreter data once per frame, an encoding unit which accepts the required photointerpreter data and converts it to a digital character which can be recognized by eye, and an analog to digital converter for the photointerpreter data (where needed) and for the heat budget data. Video sync signals and command signals are accepted as control inputs to this system; and video data and heat budget data are accepted as data inputs.

The necessary data handling equipment for the system as described, with analog transmission of video data, will weigh 3 lb. Its volume will be 50 cu in. and it will require 2 W of power.

b. Command Data

The data handling requirements for the command link are listed in Table 2-2 of Volume 5. The list is representative of a system to meet the requirements of the SMS vehicle. It is given as an aid in determining the command data handling requirements.

There are about 100 different commands to be encoded and transmitted via this system. Eight bits have been reserved for this purpose so that there can ultimately be 256 commands given in the word structure.

The command data handling system for the medium capability vehicle has provision for command verification. This involves storing the command and transmitting it to the ground over the telemetry link. A return signal over the command link is then sent as an "Execute" command, if the command has been verified.

A discussion of PCM encoding or tone digital encoding is given in the section on telemetry data handling in Section 2. B of Volume 5.

The equipment involved in the data handling consists of signal processing equipment for synchronizing followed by a decoding matrix which determines the interpretation of the command and routes it to the proper actuator for implementation. The input to this equipment is the demodulated signal from the command receiver.

The estimates of size and weight for the data handling system are based on using a tone digital command link during the launch and ascent to orbit and a PCM system when on station. The dual system is used because the world-wide tracking net is completely equipped only with the tone digital equipment. Once on-station the SMS requires a capability for additional commands; and, therefore, a PCM command system (as used at OGO stations) will be used at the SMS command and data acquisition station.

c. Telemetry Data

An itemized list of telemetry requirements for the medium capability satellite is given in Table 2-3 of Volume 5. The points to be monitored are chosen as being representative of a typical system for the vehicle requirements, and are presented as an aid to determining the telemetry requirements.

There are about 100 points to be monitored and the various frequency and precision requirements bring the data rate to about 200 bits per sec. Ten pulses every 1/4 sec are reserved in the format to carry a command back to the operator for verification.

The data processing equipment for the telemetry system arranges the data for presentation to the telemetry transmitter. It consists of a multiplexing unit which switches in various points to be monitored in a predetermined sequence, a signal conditioner which normalizes various voltage ranges to a most suitable range for processing, and an analog-to-digital converter to encode the voltage to a digital code for transmission.

It is estimated that the equipment to meet the telemetry data handling requirements will weigh 17 lb. occupy 663 cu. in. and consume 17.1 W. Portions of the high rate section in the Nimbus telemetry are compatible with this data rate. However, the multiplex unit will require redesign to the high frequency rate.

4. Control Subsystem

The Atlas-Agena is used to boost the medium capability satellite into orbit. The Agena is first placed in a parking orbit after launch and then fired into the transfer ellipse at the proper time. After burnout, the Agena is aligned with respect to inertial space to provide the correct orientation for the apogee motor impulse. The SMS payload is then spun-up on the Agena spin table to an angular rate of about 100 RPM. This value is chosen to keep the disturbance torques encountered in the transfer ellipse from causing the vehicle's inertial orientation from deviating significantly from the desired alignment. Spinning the satellite nullifies the effect of unbalanced torque caused by apogee motor misalignment. The maximum torque capability of the attitude control system is 0.25 in.-lb while the unbalanced torque produced during apogee burn is expected to be 100 in.-lb. Increasing the control system torque capability to this level is not desirable because of the effect on other components in the system. Providing 3-axis stabilization during apogee burn causes additional requirements to be placed on the attitude sensors, because the satellite's orientation is not the same as during synchronous orbit, and, therefore, is not recommended.

After apogee burn, passive despin is used to remove most of the spin rate. It was chosen as the means of despinning the vehicle because of its simplicity.

The medium capability satellite has a 3-axis control system as shown in Figure 2-8. Its first use is after passive despin to control the vehicle rates in yaw, pitch, and roll to 0.1, 0.1 and 0°/sec, respectively. This is accomplished by the use of rate gyros and gas jets. After these rates are established the solar paddles are extended. The Sun is then acquired by the 180° Sun sensor mounted on the solar paddles (which are still locked in a plane normal to the roll axis). As soon as the Sun is detected, yaw and pitch signals are sent to the gas jets to align the paddles normal to the Sun line. When the alignment is complete, the vehicle is caused to roll about the Sun line at 0.1°/sec until the Earth is detected by the horizon sensor. The detection signal will unlock the solar paddles. This permits the Sun sensor to control the paddles normal to the Sun line in pitch only. The roll and pitch reaction wheels will be activated to permit acquisition of the Earth. After Earth acquisition, the constant speed wheel will be powered to provide yaw orientation by gyrocompassing. When the satellite is fully stabilized, the velocity error correction will be commanded from the ground; and either the four yaw and pitch jets or the one opposite jet will be operated depending on the direction of the correction. The 3-axis nozzle geometry is shown in Figure 2-9. Because the pitch jets are used for unloading the pitch reaction wheel, station keeping, and velocity error correction, a priority logic circuit is required to guarantee that the unloading function will be performed when required. The station keeping ground command will only operate the four yaw and pitch jets, because the correction is always in one direction.

In the normal tracking mode, roll and pitch errors are determined by the horizon sensors; and corrective control torques are produced by the variable speed wheels. In the yaw axis, the constant speed wheel senses yaw errors and produces corrective torques. The gas jets are used for unloading the pitch and roll reaction wheel, and for station keeping.

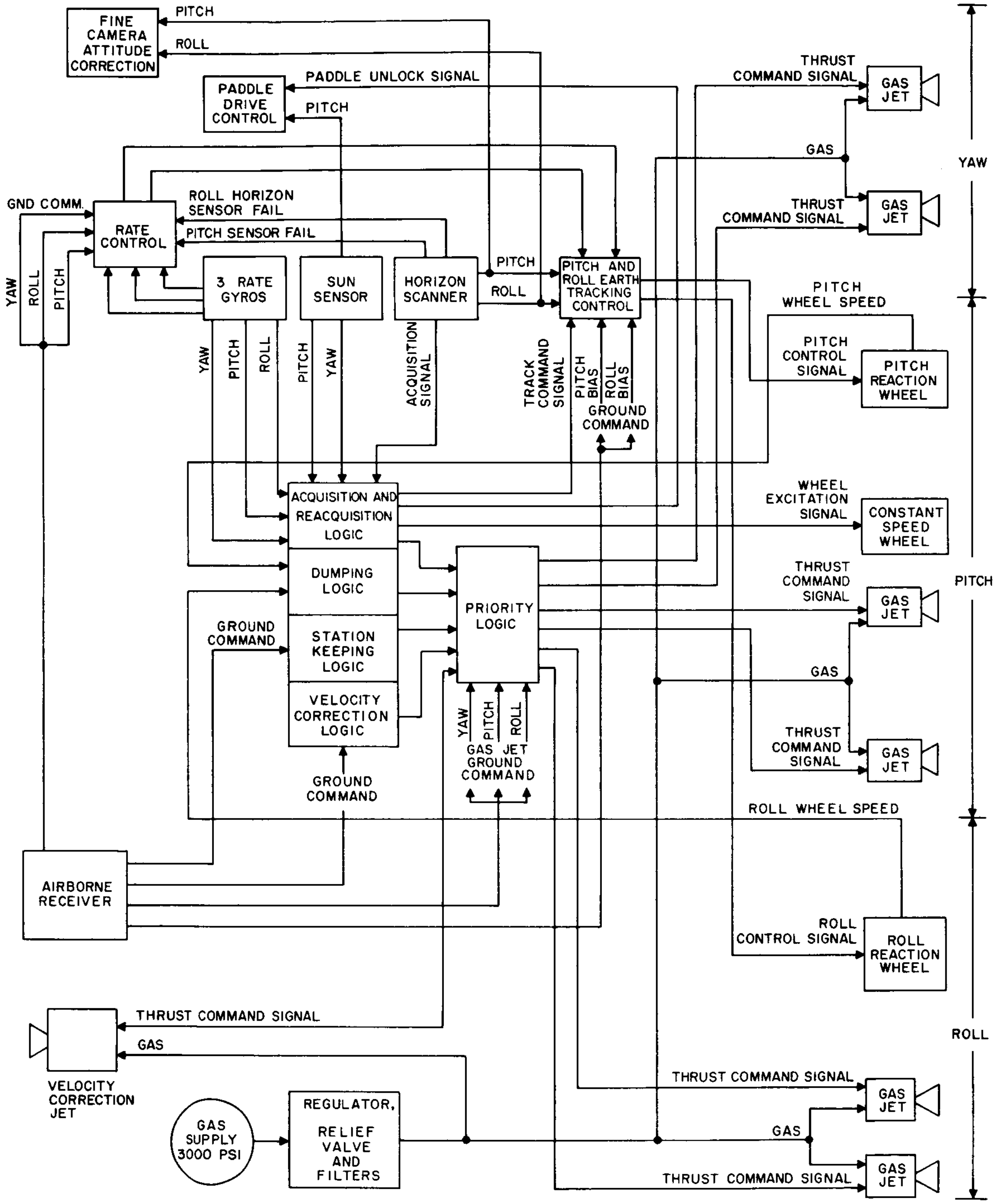


Figure 2-8. 3-Axis Stabilized Control System

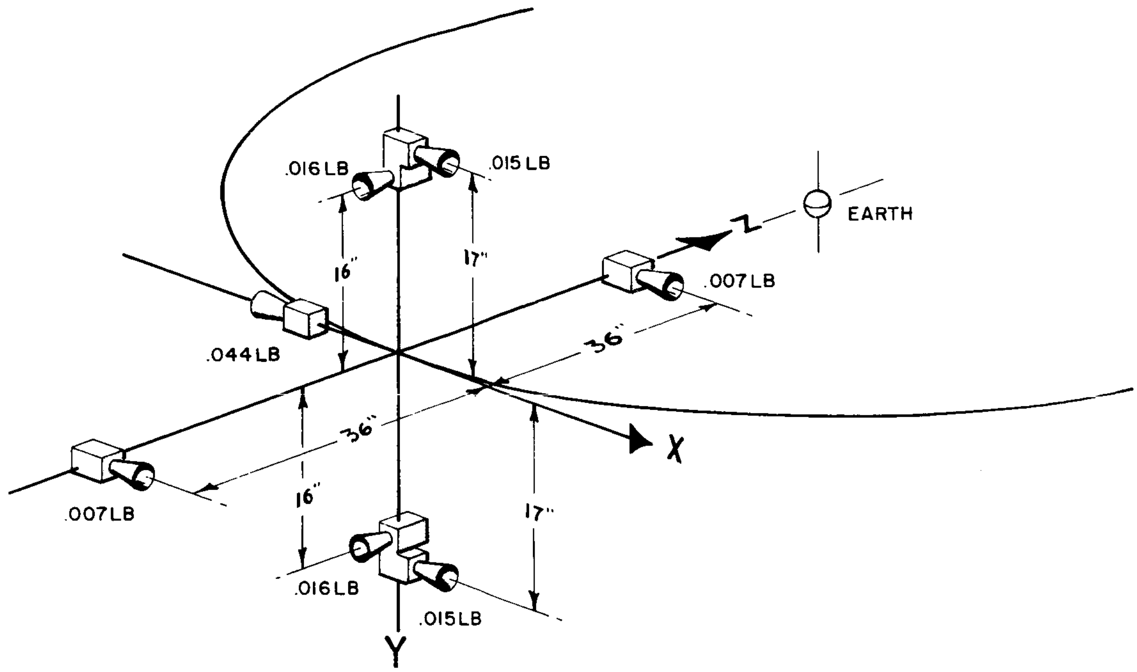


Figure 2-9. Nozzle Geometry for Attitude Control Gas System

Manual control of the 3-axis control system, by means of ground command, has been provided. Both velocity error and station keeping corrections are activated only by ground command. The three sets of attitude control jets, which are normally operated automatically, can be manually controlled by ground command. This control provides a manual backup for unloading the wheels and, also, could be used for acquisition in case of failure in the automatic system.

Ground command bias signals can be provided in pitch and roll. These signals will be used to correct steady state errors in the horizon sensor or alignment errors in the meteorological sensors. As a backup for the horizon sensor and the gyro-compass, ground command can activate the rate gyros to control the satellite in a rate mode. In this mode of control, satellite rates can be commanded in three axes. Attitude information obtained on the ground from the full Earth pictures can then be used to control the satellite's attitude in the rate mode.

A high pressure cold gas system is used because the total impulse is not large enough to realize the advantages of the higher specific impulse of a hot gas system. From a reliability point of view, the advantages of a single cold gas system of proven techniques far outweigh any weight savings that might be obtained by using either hot gas or a combination of hot and cold gas systems.

The gyrocompass system provides spacecraft stabilization rates of 10^{-3} °/sec when external disturbances, and horizon sensor effects are included. Basic pointing accuracy achievable with this system is better than 0.1° for disturbance periods where normal solar activity exists. This pointing accuracy will necessarily be degraded by horizon sensor accuracy characteristics. Thus a 1° sensor will degrade pointing accuracy to about 1°. The basic pointing accuracy is also degraded during periods of solar flare. The extreme solar gusts with magnitudes 75 times normal solar pressure, will cause pointing errors of about 1°. This accuracy is compatible with the contemplated sensor system.

5. Power Supply

The medium capability satellite will use solar cells and nickel cadmium batteries for its normal power supply, with silver zinc "one-shot" batteries supplying power for those systems used during ascent to synchronous orbit. Analysis has shown that two sets of batteries results in lower system weight. Solar paddles rotate about the pitch axis, when deployed, to present maximum surface area to the Sun at all times.

To attain the lowest power system weight, it was necessary to limit the operation of the sensor equipment during the occult period when the satellite is receiving no Sun power. It is not believed that this will impair mission effectivity because it will affect only 2 cycles per day at the worst period. These periods occur twice a year at equinox. For the purpose of this and following configuration discussions, the term "limited" is applied to power supplies that are not capable of full normal operation during occult periods.

A further limitation was placed on the system in that all power consuming sensor equipment cannot be used simultaneously. The cumulative effect of collecting, processing, and transmitting information from all sensors simultaneously results in peak power demands requiring an excessively large power system. Hence, information acquisition and transmission has been programmed so that a more reasonable power supply suffices. The power required from the solar paddles represents the average wattage over 24 hours.

This average wattage of 272.2 results in a solar paddle area of approximately 43 ft^2, based on 7.5 W/ft^2. This figure takes into account the angle of incidence of the Sun as it deviates 23.5° from the normal to the paddle surface during the year. The solar array has been sized at 50 ft^2, allowing approximately 18% margin for cell degradation and random damage. Solar cell efficiency was taken as 6%.

Table 2-9 presents a detailed breakdown of the subsystem power demands. For discussion of the factors used in power system design see Volume 5.

TABLE 2-9
SUBSYSTEM POWER REQUIREMENTS - MEDIUM CAPABILITY SATELLITE

Item	Acquisition Power (W)	Track Average Power (W)	Track Peak Power (W)
1. Attitude Control and Station Keeping			
Injection Correction	24 W for 6 hr	0	0
Cold Gas System	24 W for 1 hr	24	24
Reaction Wheels (2)	0	9	18
Constant Speed Wheel (1)	0	18	36
Horizon Sensor (2)	0	4	4
Sun Sensor	0	0	0
Rate Gyros (3)	42 W for 1 hr	24	84
Electronic Control Assy (1)	13 W for 1 hr	26	75
Total	79 W	105	241
2. Communications			
Telemetry Data Transmitter (2)	15 W for 6 hr	15	15
Sensor Data Transmitter (2)	40 W for 6 hr	55	55
Relay Receiver (2)	3 W for 6 hr	3	3
Command Receiver (2)	2 W for 6 hr	2	2
Antenna System	0 W for 6 hr	0	0
Total	60 W	75	75
3. Data Handling			
Telemetry (1)	17.6 W for 6 hr	17.6	17.6
Sensor (1)	0 W for 6 hr	2	2
Command (1)	11.5 W for 6 hr	11.5	11.5
Programming	2.0 W for 6 hr	2	2
Total	31.1 W	33.1	33.1
4. Power Supply			
Solar Paddles Drive and Charge Regulator	7.0 W for 6 hr	7	7
Total	7.0 W	7	7

TABLE 2-9 (Continued)

Item	Acquisition Power (W)	Track Average Power (W)	Track Peak Power (W)
5. Sensor Equipment *			
Low Resolution, Full Earth Disc System	0	7	15
Camera and Aiming Adjustments	0	4.8	6
High Resolution TV Camera	0	47	50
High Resolution IR Earth Heat Budget**	10	18.3	60
Total	10 W	77.1	131

* If the use of sensor equipment is staggered (i.e., IR heat budget and high resolution vidicon camera are not in simultaneous operation) then the peak power may be reduced. The summation will then by 15 + 3 + 3 + 50 + 10 = 81 W rather than the 131 W shown.

** During the 30 minute data acquisition cycle, the IR equipment uses 60 W for 5 minutes and 10 W for 25 minutes, giving an average of 18.3 W for the cycle.

6. Thermal Control

The 3-axis stabilized spacecraft has a heterothermal structure (see Figure 2-10). However, total stabilization has one advantage as far as thermal control is concerned which a spin stabilized system cannot offer. This advantage is the application of the shadow box technique discussed in Volume 5. Briefly, this technique involves mounting equipment on those sides of the spacecraft which always face space and protecting these areas from any incident Sun rays by a deep grid. Because this technique of thermal control cannot be applied to all parts of the vehicle, other methods of thermal control are required. In the 3-axis stabilized spacecraft thermal control will be achieved by means of the shadow box, thermal storage, local heating elements, and insulation.

The components to be controlled by the shadow boxes, and their heat dissipations, are listed below (see Figure 2-2).

a. Upper Shadow Box

Left Side Component	Left Side Heat Dissipation (W)	Left Side Temperature Limits (°C)	Right Side Component	Right Side Heat Dissipation (W)	Right Side Temperature Limits (°C)
Battery	25	25±10	Battery	25	25±10
Battery	25	25±10	Battery	25	25±10

	Left Side			Right Side	
Component	Heat Dissipation(W)	Temperature Limits(°C)	Component	Heat Dissipation(W)	Temperature Limits(°C)
Attitude control	26	25±10	Telemetry	8.5	25±10
Command receiver	11.5	25±10	Telemetry	8.5	25±10
Data programmer	2	25±10	Relay receiver	3	25±10
Telemetry command	1	25±10	Telemetry command	1	25±10
Total	90.5		Total	71	

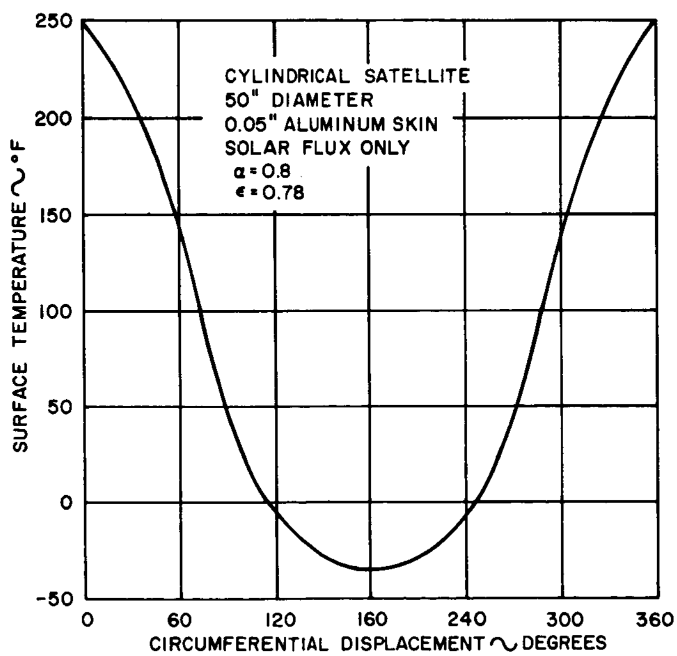


Figure 2-10. Surface Temperature Distribution

The area of each upper shadow box is 2.78 ft^2. The average power density of each side is 29 W/ft^2. Allowable temperature range on all the equipment is 25±10°C (77±18°F). To keep the surface temperature at 77°F, an emissivity of 0.69 is needed. As pointed out in Section 4 of Volume 5 the solar absorptivity does not affect the control surface of the shadow box.

The control surface material (bottom of the shadow box) will be 1199 aluminum, mechanically polished with Brytal electropolish (15 min anodize in 15% H_2SO_4 solution). This material has been tested at the Engineering Radiation Laboratory, UCLA. It has an emissivity of 0.74 (α_S = 0.15). This will give a shadow box control surface temperature of 67°F. This allows enough temperature gradient between the surface and the equipment for efficient heat dissipation. It should be remembered that the equipment controlled by the shadow box is insulated-from the rest of the craft. The equipment temperature, therefore, will be only a function of the control surface temperature and the temperature gradient allowed by the available heat transfer path.

b. Lower Shadow Box

Left Side			Right Side		
Component	Heat Dissipation(W)	Temperature Limits(°C)	Component	Heat Dissipation(W)	Temperature Limits(°C)
Sensor data transmitter	90	25±10	Relay receivers	6	25±10
High resolution camera	25	40±5	High resolution camera	25	40±5
Heat budget sensor	60	25±10	Low resolution camera	10	25±10
Horizon scanner	4	25±10	Horizon scanner	4	25±10
Total	179		Total	45	

The component in the high resolution camera requiring 40±5°C is the image orthicon tube. Because the control temperature (25°C) is lower than the allowed temperature for the orthicon tube, a thermostatically controlled surface heating element will be used to maintain the orthicon temperature at 40±5°C. The orthicon tube weighs approximately 1 lb and has a surface area of 0.6 ft^2. The estimated heating element power requirements will be below 3 W.(See discussion on surface heating elements in Volume 5.)

The area of each of the lower shadow boxes is 6.7 ft^2 giving a power density of 26.7 W/ft^2 for the left side and 6.71 W/ft^2 for the right side. To maintain a 25°C temperature, an emissivity of 0.635 is required for the left side and 0.16 for the right side. The control surface base material for the left side will be 1199 aluminum with a 0.001 in. thick B-72 Acryloid coating having an ϵ of 0.6 and an α_S of 0.2. This material was tested at the Los Angeles Division of North American Aviation.

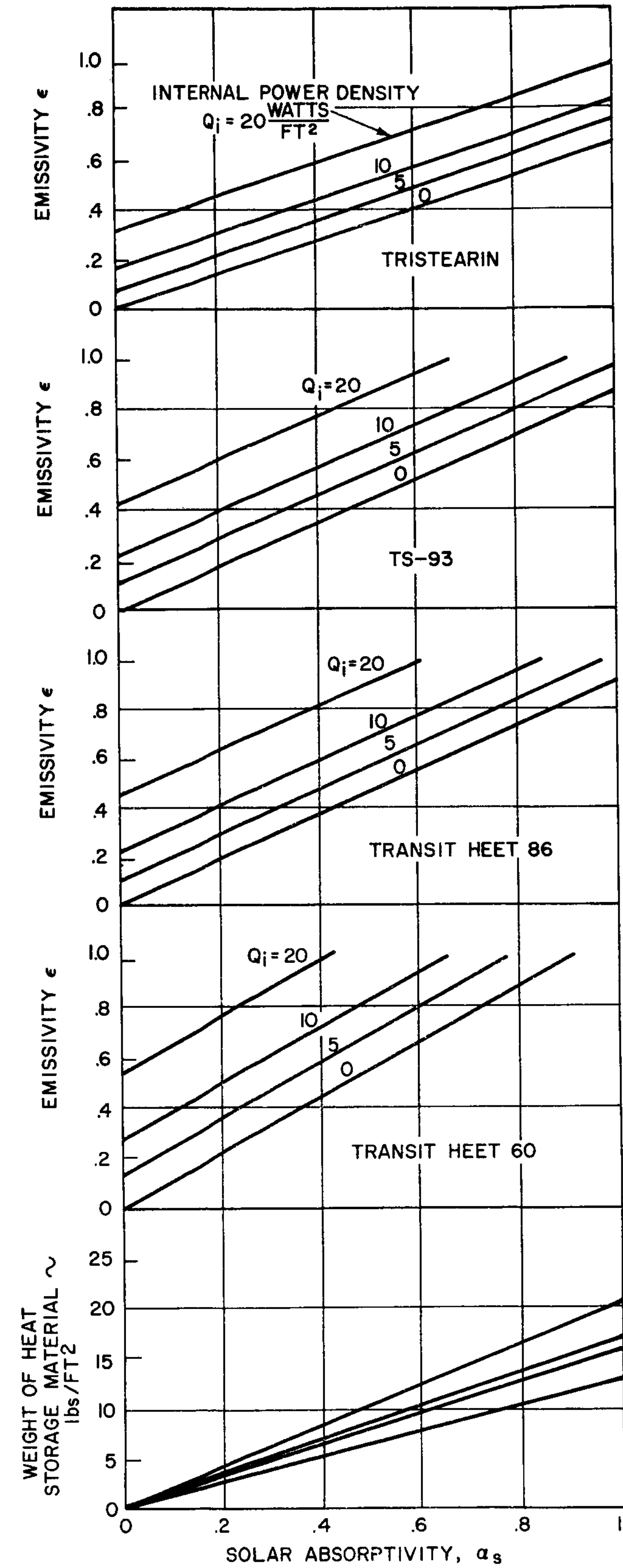
The control surface (bottom of shadow box) material for the right side will be rough polished aluminum. This material has an ϵ of 0.182 giving a surface temperature of 65°F with a wide margin for equipment to surface temperature gradient. The shadow box perimeter frame (for all 4 boxes) outside surface will be painted with zinc sulfide-silicone, ZW60, produced and tested by Jet Propulsion Laboratories. This paint has an ϵ of 0.91 and an α between 0.21 and 0.29 after 0.29 after 4000 hours of solar vacuum radiation exposure. The inside surface of the frame will be polished aluminum with an ϵ of 0.0.03 to 0.06 and an α_s of 0.2. The high emissivity of the frame outer surface and the low emissivity of the inside surface are essential to the shadow box principle. See Section 4 of Volume 5 for the advantages of the shadow box technique..

c. Components to be Controlled by Heat Storage Material

The components that are to be controlled by heat-storage material are listed below:

Component	Heat Dissipation (W)	Temperature Limits (°C)
3 reaction wheels	15	25±10
3 gyros	5	25±10
Regulator and inverter	3.6	25±10
Total	23.6	

To reduce the amount of heat-storage material, the components that are favorably grouped together will be compartmentized and insulated. Each compartment will have its own heat storage material core - radiation panel. The amount of heat storage material and the radiator coatings required for each panel will be determined by the power density of the components in each compartment. These relationships are shown in Figure 2-11 for a flat plate and in Figure 2-12 for a cylindrical surface. As seen in the two figures, the weight of a thermal storage temperature control system is primarily a function of α_s. To reduce the temperature control system weight it is important, therefore, to have as low an α_s as possible. The corresponding emissivities are dictated by the power density of a given panel, as shown in these figures. At present, the development of low absorptivitity surface coatings that are not degraded under ultraviolet exposure is being carried out at a number of research and development institutions. Figure 2-13 shows the behavior of a low α_s material under exposure conditions that approach the required SMS endurance. These data are from actual tests. The development of a paint with an α_s of 0.1 ($\epsilon = 0.8$) has been recently reported by the Corning Research Laboratory. The pigment consists of fine particles of ultrapure fused silica, and the paint vehicle is an experimental silicone varnish that evaporates at elevated temperatures, leaving a matrix of fine particles of fused silica with good adhesion. This coating formulation is presently undergoing ultraviolet vacuum exposure tests. It is expected that the coating will be very stable because its only ingredient, the ultrapure fused silica, is transparent to the ultraviolet spectrum above 2000Å. The fused silica is also very resistant to particle radiation (see Tables 4-1, 4-2, and 4-3 in Volume 5 for radiative properties of other materials.)

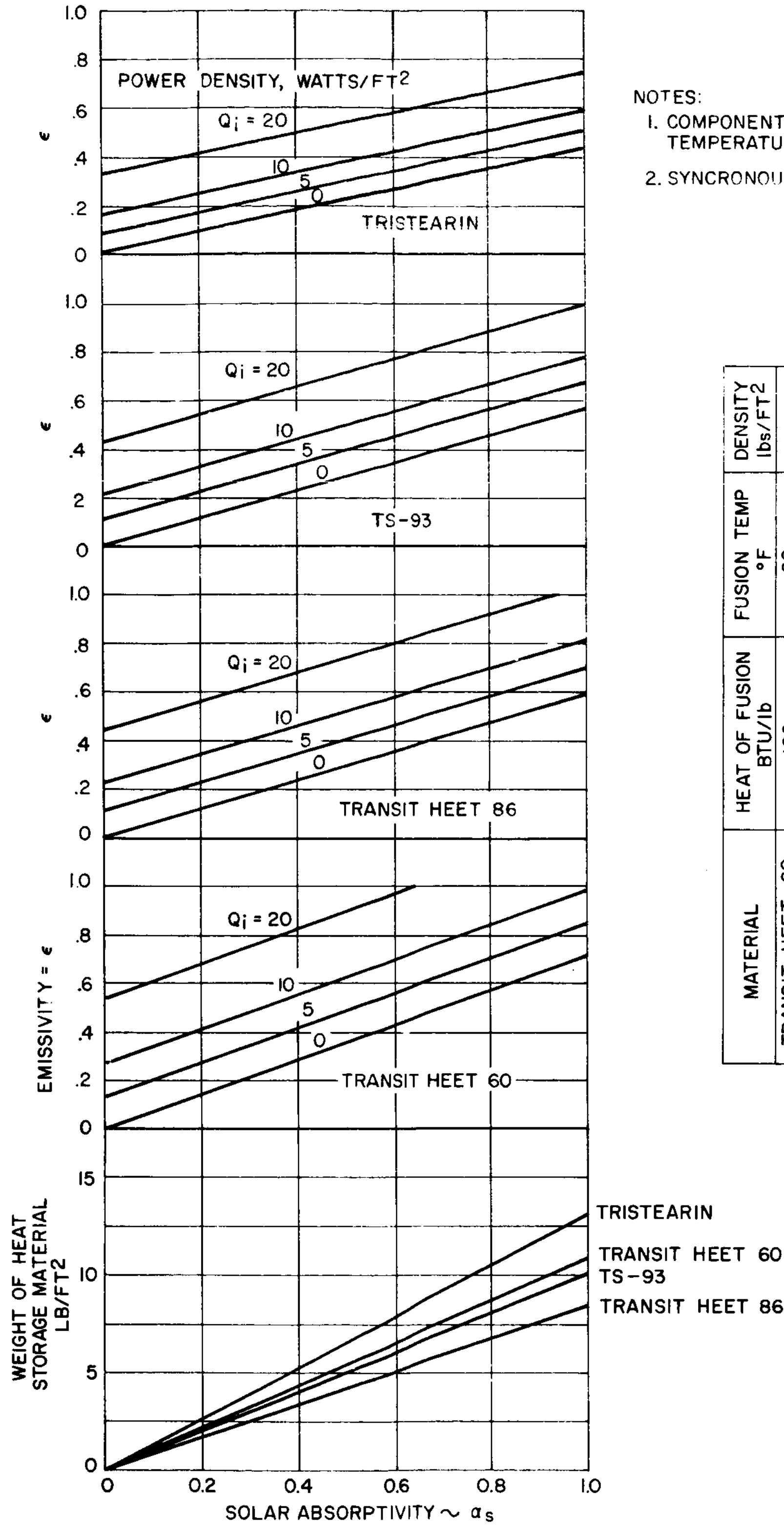


NOTES:

1. COMPONENT TEMPERATURE ≈ FUSION TEMPERATURE.
2. SYNCHRONOUS ORBIT CONDITIONS.
3. FLAT SURFACE. SEE VOL. 2, SECTION 2 FOR CYLINDRICAL SURFACE.

MATERIAL	HEAT OF FUSION BTU/lb	FUSION TEMP °F	DENSITY lbs/FT2
TRANSIT HEET 60	100	60	100
TRANSIT HEET 86	130	86	100
TS-93	107	93	—
TRISTEARIN	82.1	133	—

Figure 2-11. Performance of Thermal Storage Temperature Control System - Flat Plate



NOTES:
1. COMPONENT TEMPERATURE ≈ FUSION TEMPERATURE.
2. SYNCRONOUS ORBIT CONDITIONS.

MATERIAL	HEAT OF FUSION BTU/lb	FUSION TEMP °F	DENSITY lbs/FT2
TRANSIT HEET 60	100	60	100
TRANSIT HEET 86	130	86	100
TS-93	107	93	—
TRISTEARIN	82.1	133	—

Figure 2-12. Performance of Thermal Storage Temperature Control System-Cylindrical Surface

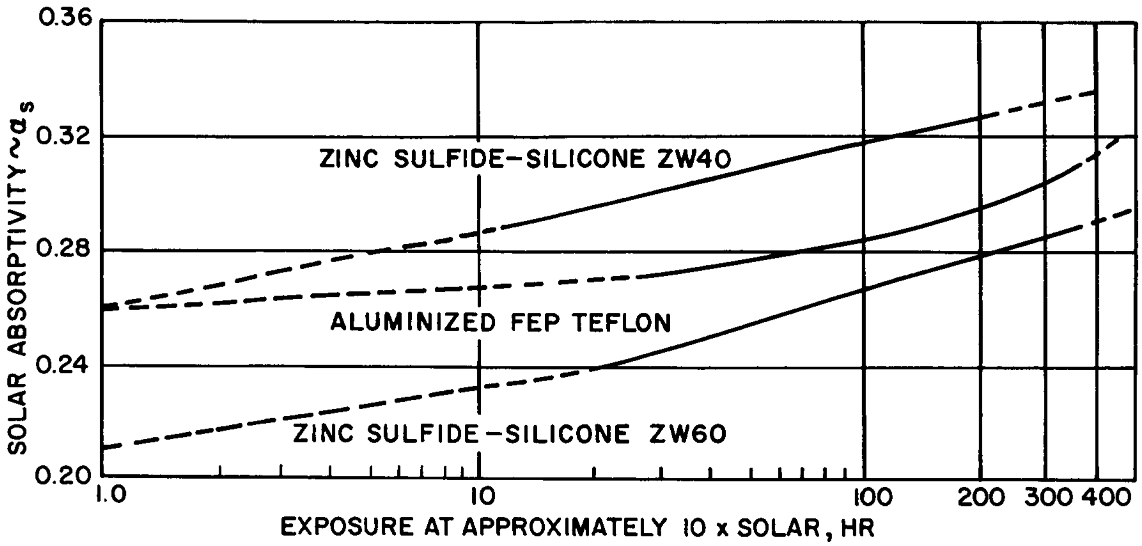


Figure 2-13. Degradation of Thermal Control Surfaces in a Vacuum Due to Ultraviolet Rays

7. Structure and Materials

a. Design Considerations

The launch and orbit injection environments determine configuration and structural design to a great extent. It is assumed that prelaunch environment heat, dust, fog, temperature, and transportation shocks can be adequately controlled.

During the launch phase, broadband vibration excitations (generated at booster ignition and during ascent, and transmitted to the payload aerodynamically and mechanically) represent the most critical conditions encountered. The magnitude of vibration varies during launch in amplitude and excursions. The time of peak severity is common to all launch vehicle spacecraft configurations and generally identified as:

(1) Some few seconds following initial ignition

(2) During transonic flight

(3) At the time of maximum dynamic pressure

(4) Toward stage burnout

The excitation is generally random in character with frequencies from 5 to 3000 cps. In addition, some liquid fuel boosters exhibit low longitudinal (below 53 cps) excitations due to fuel flow pulsations and burning instabilities.

Basic design considerations to ensure adequate structural strength in the face of the severe vibration spectrum include the following:

(1) Design supporting structure such that resonating frequencies of the installed equipment do not coincide with the critical frequency ranges occurring during launch or orbit achievement.

(2) Where possible, design by the octave rule in which the natural frequency of the individual component differs from the frequency of its mounting by a factor of two.

(3) Design equipment support structure so that transmissibility factors of 10 or lower are achieved. Spacecraft structures have exhibited isolated amplifications in excess of 30.

(4) Mechanical vibration isolators should not bottom out during maximum transient accelerations. Added damping should be provided by slip damping of joints where possible.

(5) Design individual equipment mountings for minimum dynamic cross coupling response.

(6) Mount all equipment so that their mass cg falls as close to mounting attachments as possible.

(7) Locate equipment packages as uniformly as possible throughout the spacecraft and arrange the basic structure so that little change occurs to the overall dynamic response with equipment changes.

The basic spacecraft configuration (Figure 2-14) comprises two geometrically dissimilar sections. The lower section consists of a hollow double skinned cylindrical structure housing the apogee motor, telemetry, power supply, attitude control, passive despin weights, and solar paddle supports. The booster end of this section is attached to a conical adapter skirt by means of a circular Marman clamp, engaging on the periphery of an end closure frame and the movable portion of a booster spin table. The opposite end of the section terminates in a bulkhead frame. The sensors and various other optical equipment are housed in front of this frame.

The circular shape was chosen over other possible configurations because:

(1) It permits maximum efficient structural continuity between the spacecraft and booster adapter and using known and tested separation clamps.

(2) It permits using curved external skin panels that have higher resistance to acoustic fatigue than flat panels.

(3) It provides maximum stiffness versus weight for overall housing structure and permits a simpler structure for internal stiffener rings and longerons.

(4) The circular section results in maximum usable space for equipment as limited by the envelope formed by the folded solar panels and the apogee motor.

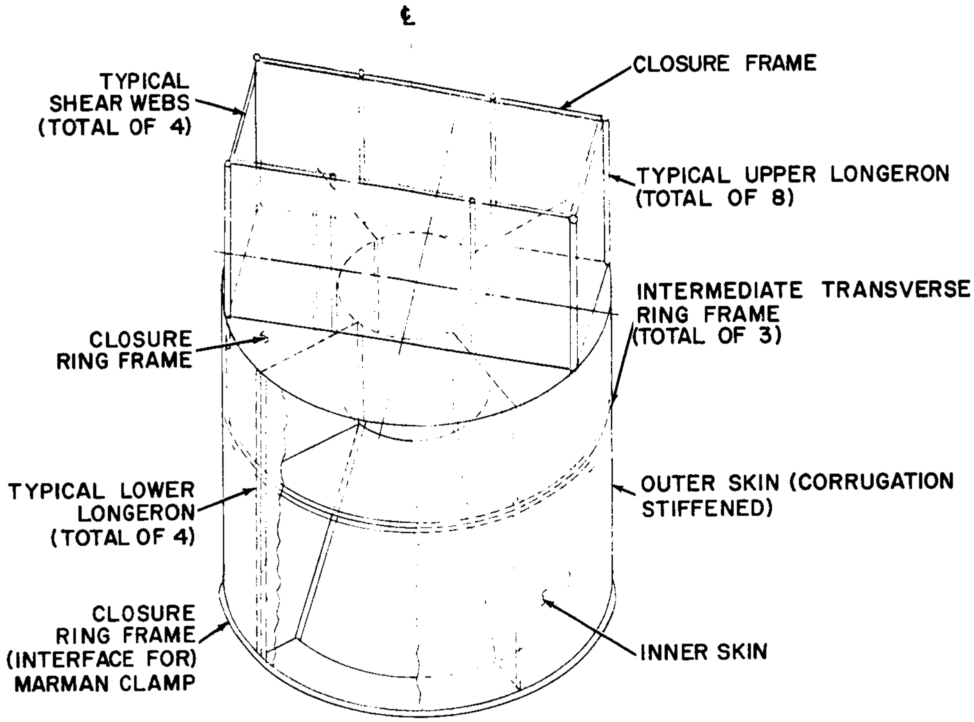


Figure 2-14. Basic Structural Arrangement - Medium Capability Spacecraft

As shown in Figure 2-14 the structural framework of the lower spacecraft section consists of a thin-walled stiffened outer and inner cylindrical shell (or skin), four full-length longerons and a number of transverse circular shelves or decks of substantial depth. The latter fit between the longerons and divide the housing structure into four bays. At the two ends of the housing, the shelves form complete ring frames. Each bay is designed to support a particular array of equipment and is provided with a number of radial ribs and transverse beams between the shelves and longerons to support individual pieces of equipment. To simplify the inclusion of shadow boxes for thermal control (Figure 2-2), the curved outer skin is locally interrupted in two diametrically opposed sections of the two intermediate bays and replaced with flat structural inner beams for mounting equipment. Though not shown in Figure 2-2, suitable stiffening and back-up structure would be provided for support of individual equipment items. The arrangement of the structure outlined provides for maximum load continuity and a good degree of rigidity along three mutually perpendicular axes. This is essential to ensure low dynamic response to shock and vibratory loads. In addition, the structural grid work proposed is well adapted to mounting a wide variety of equipment by inclusion of additional bulkheads, ring frames, and intercostals, as required. The upper section of the spacecraft houses sensors, cameras, and other optical equipment. The basic structure is a full length rectangular box beam, the corners of which are made up of longeron members.

The sides of the structure are made up of support beams upon which equipment is mounted. Local stringers, transverse frames, and shelves would also be provided where necessary. Passive thermal control of the box beam housing is achieved by shadow boxes mounted on two opposite sides of the beam. The design of these shadow boxes is similar to those located in the lower spacecraft section. As conceived, the boxes would be designed as nonstructural. However, their construction would have to incorporate sufficient rigidity to survive the vibratory environment of launch.

b. Weight Summary

Satellite weight is itemized in Table 2-10. The weight shown as present design is one in which known techniques are applied to the design. A comparative list itemizes the weight reductions possible by developing certain items such as subliming solids (as described in Appendix A). Figure 2-15 shows the cg travel from launch until synchronous orbit injection.

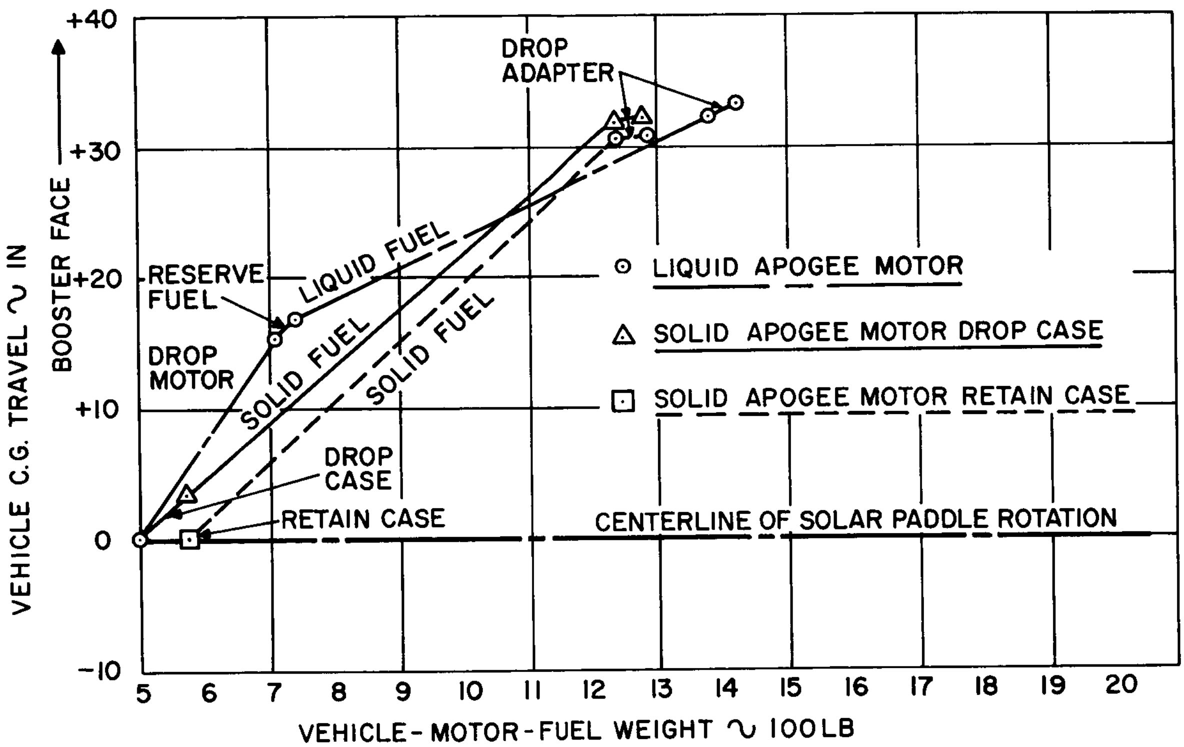


Figure 2-15. CG Travel - Parking to Synchronous Orbit

c. Load Paths

Support fittings for the majority of equipment contained in the lower section (Figure 2-14) would be designed to transmit their inertia loads to the shelves, longerons, ribs, and transverse beams provided. The shelves then transfer their lateral loads as shear flows to the external and internal skins, and the ribs and beams, in turn, transfer their longitudinal loads to the longerons.

TABLE 2-10
WEIGHT SUMMARY - MEDIUM CAPABILITY SATELLITE

Item	Present Configuration Weight (lb)	Future Development Weight (lb) (Appendix A)
Sensor Equipment Total	83.0	83.0
Low-resolution, Full Earth, Disc System	10.0	10.0
Includes: Sensor Image Lens, Shutter, Iris, Vidicon Camera System, Power Supply, Amplifier, Sync, AFC, etc.		
High resolution, Narrow Field of View System	20	20
Includes: Primary Reflecting Optical System, Filter Wheel, Shutter		
Scanning Mirror	8.0	8.0
Includes: Mirror, X and Y Drive Motor, Gimbaling System		
High resolution TV Camera	15.0	15.0
Includes: Image Orthicon Camera Systems, Power Supply, Amplifier, Sync, AEC, etc.		
High resolution Earth Heat Budget	30.0	30.0
Includes: Lens, Thermistor Bolometer Detector System, Power Supply, Sync, Shutter, etc.		

TABLE 2-10
WEIGHT SUMMARY - MEDIUM CAPABILITY SATELLITE (Cont'd)

Item	Present Configuration Weight (lb)	Future Development Weight (lb) (Appendix A)
Communications Total	22.0	22.0
Telemetry Data Transmitter (1)	1.3	1.3
Sensor Data Transmitter (1)		
Includes TWT	1.7	1.7
High Voltage Power Supply	6.0	6.0
RF Driver of Modulation	2.0	2.0
Relay Receiver (1)	3.0	3.0
Command Receiver (2)	3.5	3.5
Antenna, Telemetry (4)	0.5	0.5
Slot (1)	0.5	0.5
Sensor (1)	3.5	3.5
Data Handling Total	29.5	29.5
Telemetry (1)	17.0	17.0
Sensor (1)	3.0	3.0
Command (1)	9.0	9.0
Programming	0.5	0.5
Power Supply Total	127.7	101.7
Batteries (Ni-Cad)	39.7	39.7
Batteries, Solar Paddles Selector, Charger, Regulator	12.0	12.0
Solar Paddles (Conventional) (Flexible)	76.0	50.0

TABLE 2-10
WEIGHT SUMMARY - MEDIUM CAPABILITY SATELLITE (Cont'd)

Item	Present Configuration Weight (lb)	Future Development Weight (lb) (Appendix A)
Attitude control and station keeping	103	61.8
Cold Gas System		
(Including orbit injection corrections)		
Bottles (4) (Empty)	34	
Gas (Nitrogen)	24	
Tanks (2) (Empty)		0.8
Subliming Solid		16.5
Nozzles, Valves, Regulators, Lines, etc.	2	1.5
Reaction Wheels (2)	10	10
Constant Speed Wheel (1)	10	10
Horizon Sensor (2)	8	8
Sun Sensor (2)	0.4	0.4
Rate Gyros (3)	2.6	2.6
Electronic Control Assembly	12	12
Wiring and Harness	22	22
Thermal Control	10	10
Structure (Using Marman Clamp Separation)	108	
(Using Shaped Charge Separation)		88.3
Despin Mechanism	18.8	18.8
Satellite Weight	524	437.1
+ Motor Case	75	70
+ Adapter	42	42
+ Propellant	670	572
Total from Booster: (Payload at Separation)	1311	1121.1

Because of the low loading index associated with this type of spacecraft, the skins should consist of a composite of light corrugated stiffeners with an external outer skin for thermal control. The corrugated skins provide for rigid shell structure and efficient shear carrying material. The four longerons (consisting of built-up beams of varying depth) symmetrically placed between the two skins are designed to transmit the longitudinal inertia forces applied directly to them as well as those induced by lateral bending of the overall shell structure. The longerons and shelves also assist in stabilizing the external and internal skins. The continuous ring frames serve to maintain their circularity. Bending moments induced within each local bay area are reacted by the adjoining shelves and thence to the two skins.

All the longitudinal, lateral, and bending loads developed within the circular section of the spacecraft (Figure 2-14) are transferred to the adapter section below via the lower bay area. Longitudinal and bending loads from the longerons are transmitted to the outer skin and, thence, to the Marman clamp and spin table movable ring structure at the interface of the adapter structure and the spacecraft. Loads developed in the outer skin itself are transferred in similar fashion. Because the skin serves as the prime transfer medium for all loads applied to the adapter, added stiffness and strength would be incorporated in the lower bay area. In addition, a lower end machined ring frame is provided to stabilize the skin for local bending of the longerons and to afford a rigid back-up surface for the attachment of the Marman clamp. Therefore, with this scheme, loads are distributed uniformly to the adapter section. Inertia load distribution in the upper section would essentially follow that described for the lower section. In this case, the box beam webs would constitute the outer skin and all loads developed in this area would eventually transfer to the lower section.

As envisioned, the conical adapter shell construction would be similar to that of the spacecraft shell; a light corrugation-stiffened skin, together with a sufficient number of full length longerons and transverse ring frames. In addition, machined rings would be provided at each end for transfer of loads to the Agena-D booster interface and to the fixed support structure of the spin table.

d. Typical Equipment Support

Because the high resolution optics and camera system constitutes the largest mass of any individual piece of equipment, its method of support has been selected for illustration (see Figure 2-3).

The unit is located within the rectangular box beam. This location was selected because it provides maximum accessibility. It also isolated the camera and optics from the dynamic oscillations sustained during apogee motor firing. The primary support structure consists of three adjacent walls of the box beam. As visualized, longitudinal guide rails would be provided from the outer corner longerons to the inboard intercostals, to simplify installations. These members would also contain mount fittings and attachment points matching those of the camera unit.

Axial loads developed in the inboard intercostals would be transferred to the lower section longerons through suitable splice fittings. The loads in the corner longeron would continue on through the upper bay of the lower section, and thence be beamed through the outer skin to two of the full length longerons. Shear load transfer would be effected through the end bulkhead on the lower section. Figure 2-2 depicts the components involved. Loads imposed on the cantilevered camera mirror are carried through the fork mounting. Provisions would also be included for locking the movable mirror assembly during all phases of launch and engine firing.

e. Apogee Motor Mount

Because the weight of the apogee motor is relatively high, exceeding that of the spacecraft proper, it has been located at the adapter end to avoid the added structural weight and higher lateral amplifications which would be imposed by mounting it at the forward end. It is realized that this will require a turnabout capability for the Agena-D booster. Preliminary studies indicate that this is feasible.

The support structure for the motor consists of a thrust cone attachment to the lower end of the motor casing, with bearing pads picking up at the upper end of the casing. Axial loads in the forward direction, imparted by the motor's thrust or mass, are reacted by the bearing pads and, thence, to the four longerons. Aft loading is distributed through the thrust cone to the inner skin and end closure ring. Transverse loads would be reacted by both components.

As an alternate configuration, the mount structure for the motor might consist of a quadripod strut arrangement attached to a circular collar or ring frame fitting located at the intersection of the motor casing and nozzle (see Figure 2-16). The apex of each strut, in turn, attaches to one of the four longerons. In addition, a bearing support ring, in line with the shelf structure, is provided at the forward end of the motor casing. Thrust loads are transmitted by the four struts and lateral loads and bearing moments are distributed between the struts and forward bearing ring. To distribute lateral and aft longitudinal loads more uniformly from the struts to the motor assembly, short longitudinal pads welded to the motor casing are attached close to the inboard strut attachment points.

f. Solar Paddle Configuration and Support

The solar paddle backup structure is designed to provide adequate support for the solar cell modules mounted thereon. This structure must be rigid enough to withstand the severe vibration environment associated with the launch and powered phases of flight without sustaining damage to the solar cell electrical circuitry or loss of the cells themselves. To obtain a high flexural rigidity with minimum weight, the use of an aluminum honeycomb core sandwich is proposed for the basic paddle construction. Stiffening members would be provided along all four edges and across the center of both paddle segments. The application of a honeycomb core sandwich to the design of the paddles is considered most appropriate in view of the widespread acceptance of this method of construction to other spacecraft.

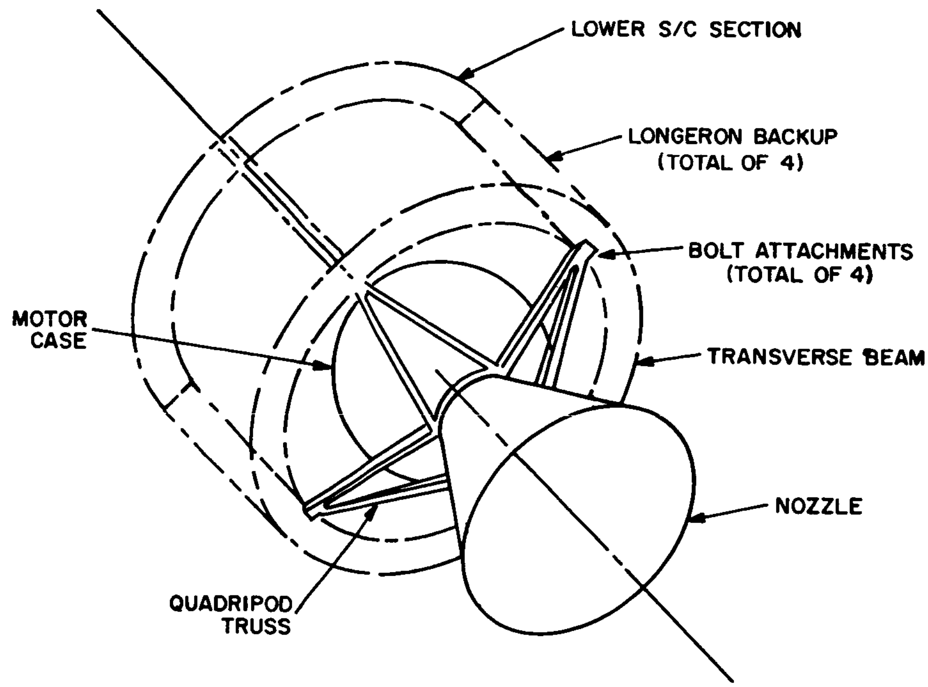


Figure 2-16. Alternate Mounting for Apogee Motor

Because the solar paddle configuration proposed is considered to be most critical from the vibration aspect, (as attested to by past test demonstrations on similar type configurations) careful attention must be placed on the support of the paddles in the stowed position. As presently conceived, the paddles are composed of two segments connected by a full length hinge running along their common longitudinal edge. In the stowed position, the outboard segments are supported at their corner points; the two outboard points are rigidly attached to the spacecraft shell structure by the squib release bolts provided; the inboard points (at the hinge line) are restrained against inward motion by a bearing support at each corner. Similar supports exist for the two outer points of the inboard segments. In turn, the inboard edges of these segments are hinged to a beam which is rigidly held at its mid-point by fittings attached to the rotating shafts. In addition to the center supports, the inboard edges are flexibly supported by the rigid edge beams attached to them. Transfer of loads from the support points to the shell structure is accomplished by the system of longerons, ribs, and shelves previously described.

Should this method of support prove inadequate in restricting amplifications and deflections of the paddle segments to tolerable levels, damping jack screws and/or predeflection of the paddles can be used.

An investigation was made of two variations of an unconventional method of paddle construction that offers promise of substantial weight reduction. Details are given in Appendix A.

One of the more important material problems concerning long time reliable satellite operation is the effect of vacuum plus radiation on joints, bearings, and lubricants. The prime friction mechanism is the cold welding of contact points that can occur because really clean metal to metal contact is possible in a vacuum. Therefore, it is necessary to have either some form of lubricant or dissimilar materials that have no cold welding tendencies.

g. Materials

The materials used to provide friction control are oils, greases, solid films, soft metallic films of low shear strength, plastics, and ceramics. All of these materials are affected in some manner by the space environment. The type of lubrication service that they can suitably provide is also affected. The materials could break down as a result of one or more of the following mechanisms: evaporative loss; creepage along ultraclean shafts; polymerization catalyzed by the clean metallic surfaces; radiation instability; wear; and absence of oxygen and other gases affecting the formation of an absorbed lubricatirg film.

It is possible to protect mechanisms from space conditions by using hermetically sealed units. However, because of the disadvantages of weight, complexity, size, and reliability, lubricants and techniques that can function in the space environment are required. As good design practice, even where hermetically sealed units seem the only feasible solution, the lubricant selected must have the lowest vapor pressure available so that, in case of puncture or seal malfunction, the system would provide lubrication for the longest possible period. Some recommended lubricants are given in Table 2-11.

Insofar as the spacecraft structure is concerned, material selection is reasonably straightforward with the major restrictions being placed on organic materials and those with appreciable outgassing characteristics. Specifically, the primary considerations in the selection of materials are strength, rigidity, thermal conductivity, fabrication, and costs. These considerations are compared in Table 2-12.

While certain steels, beryllium and titanium exhibit better strength/weight ratios than aluminum, the required gages envisioned for structural members are such that savings in weight would be small. Lack of available stocks of correct sizes and problems inherent in fastening thin gage material, as well as availability and cost, dictate that the use of these materials be confined to those structural areas of relatively high stress, or where other criteria dictate their use (e.g., fittings, attachments, bearings, etc.).

8. Propulsion

a. Apogee Motor

A solid propellant apogee motor was selected for this satellite. Calculations of weights and sizes shown in Table 2-13 indicate the weight

TABLE 2-11
RECOMMENDED LUBRICANTS FOR EQUIPMENT
EXPOSED TO SPACE ENVIRONMENT

Application	Conditions	Recommendations	Suitability Remarks
Small, lightly-loaded ball bearings	Temperature -45 to 165°F; exposed to vacuum; loads small; speeds to 8000 RPM	G-300 silicone grease with double shielded bearings	All orbital conditions
	Minimum torque essential (other conditions as stated)	F-50 silicone oil with double shielded bearings	Marginal for lifetimes beyond 6000 hr of continuous operation
	Low temperature (below that at which grease operates with an acceptably low torque) (other conditions as stated)	F-50 silicone oil with double shielded bearings	All orbital conditions
	High temperature (200°F)	Everlube 811 (sodium) silicate bonded MoS_2 film) and unshielded bearings	Speeds below 8000 RPM 480 x 10^6 cycles (unidirectional) or 165 x 10^6 cycles (bidirectional)
Fine pitch gears and gear trains	Sealed unit, temperature -45 to 165°F	MIL-G-3278; MIL-G-15793; Versilube F-30, Versilube G-300	Service tests required to establish lifetime and reliability
	Exposed to vacuum, temperature -45 to 165°F	Gears of composite materials such as sintered metals; plastics such as sintered and molded nylon, phenolic laminates, polycarbonates and polyformaldehyde	Service tests required to establish lifetime and reliability, especially where exposure to radiation is encountered

TABLE 2-11
RECOMMENDED LUBRICANTS FOR EQUIPMENT
EXPOSED TO SPACE ENVIRONMENT (Cont'd)

Application	Conditions	Recommendations	Suitability Remarks
Large, low speed support bearings	Speed below 100 RPM; temperature -45 to 165°F, exposed to vacuum	Balls or rollers of plastic such as nylon, Delvin or Lexan; when metal balls are used, lubricate with Versilube G-300 and use double-shielded bearings	Effect of radiation on plastics should be determined before they are used
	Temperature range below -45°F and above 160°F	Everlube 811 with the stated limitations or F-50 silicone oil with means of relubrication	
	Critical surfaces (for example, lenses) nearby	Everlube 811 with the stated limitations	
	No outgassing permitted (parts within evacuated and sealed electronic tubes)	Everlube 811 with the stated limitations or thin silver coatings	
	Bearings required to conduct current between races	Thin silver coating	Silver film use requires special bearing design, handling, procuring, and care of installation. Lifetime and reliability must be determined by service tests conducted on the actual application
High speed, moderate to heavily loaded gears and bearings	Temperature -45 to 165°F speed 10,000 to 3000 RPM, sealed unit	Use low volatility oil such as MIL-L-7808 or MIL-L-25336 for heavy loading in conjunction with labyrinth seals	Satisfactory for 1 year orbit at 10^{-8} mm of mercury

TABLE 2-11
RECOMMENDED LUBRICANTS FOR EQUIPMENT
EXPOSED TO SPACE ENVIRONMENT (Cont'd)

Application	Conditions	Recommendations	Suitability Remarks
Lightly loaded sliding surfaces	Temperature range -65 to 165°F exposed to vacuum	Dry lubricant such as MoS_2 or Teflon on hard metal surfaces; plastics such as nylon, phenolic laminates, and Delvin; porous metals impregnated with oil and/or MoS_2	Radiation effects on plastics should be considered before they are used. Otherwise satisfactory for 1 year orbit 10-8 mm of mercury
Heavily loaded sliding surfaces	Temperature range -65 to 200°F exposed to vacuum	Bonded solid film lubricant that is compatible with a grease or oil and a low volatility grease or oil. For example, Electrofilm 4856 with MIL-G-7118 or MIL-L-25336; bonded solid film lubricant selected for use with hard metals and high loads such as Electrofilm 77-S; dispersion of MoS_2 in grease such as Molykote Type G	Satisfactory for 1 year orbit at 10^{-8} mm of mercury
Gyroscope bearings	Require lubricants with minimum change in viscosity within operating temperature range (-45 to 165°F); sealed unit	Highly refined mineral or synthetic oil such as Teresso V-78 impregnated into a porous bearing material such as Synthane	Satisfactory for 1 year orbit

TABLE 2-12
MATERIAL SELECTION

Criteria for Selection	Material				
	Aluminum	Magnesium	Beryllium	Titanium	Stainless Steel
1. Basic density	Medium	Low	Low	Medium	High
2. Material availability, including fabrication know-how	Good	Good	Poor	Fair	Good
3. Fabrication relative simplicity	Good	Fair	Poor	Fair	Good
4. Thermal conductivity relative to thermal control requirements	High	Medium	High	Low	Low
5. Basic material cost	Low	Low	High	Medium	Low
6. Design data availability	High	High	Low	Medium	High
7. Dynamic properties (endurance limit)	Medium	Low	High	High	High
8. Engineering experience with application of material	Good	Good	Poor	Medium	Good
9. Corrosion resistance	High	Low	High	High	High
10. Handling requirements	Good	Good	Poor	Medium	Good
11. Resistance to vacuum and meteoroid penetration	High	Medium	High	Medium	High

superiority of the solid propellant motor. Investigation will show that the liquid motor and its fuel system are far more complex that the solid propellant motor.

TABLE 2-13
WEIGHT COMPARISON - SOLID VERSUS LIQUID APOGEE MOTOR

Item	Liquid	Solid
Propellant	676.0	670.0
Inert Weight		
Engine/Case	60.5	75.0
Fuel Tank	26.0	
Oxidizer Tank	33.0	
Pressurizing Nitrogen	23.0	
Plumbing, Valves, etc.	2.0	
Total	820.5	745.0

Further, the tolerances on motor weight at termination of firing, or burnout, are very close on the solid motor case, but range from 4 to 7% of fuel weight on the liquid motor. The uncertainty of final satellite weight and center of mass would probably dictate the separation of a liquid motor after orbit achievement, because the gas penalty to stabilize an out of balance satellite is significant. Figure 2-17 and Figure 2-18 illustrate the estimated weight of apogee motors required for the range of satellites under investigation.

Estimates were made for the solid propellant motor capable of supplying a Δv of 6030 sec. The proposed design would consist of a spherical propellant chamber and would use propellant having an I_{sp} = 290 sec in a vacuum. The motor would have a thrust of 4200 lb and burning time of 47 sec, resulting in an initial acceleration of 3.0 g with a maximum acceleration of approximately 5.8 g. Propellant fraction is assumed to be 0.90 and tolerance on total impulse ±1%.

b. Launch Vehicle

The target weight of the satellite was set at 500 pounds to ensure that it could be placed in orbit by the Atlas-Agena booster. The target weight has been closely approximated.

The Atlas-Agena consists of two stages: the Atlas-D, a liquid fueled rocket rated at 367,000 lb of thrust at sea level; and the Agena-B, a liquid fueled restartable rocket with rated thrust of 16,000 lb. It is conservatively estimated that this booster can place 1430 lb in a 22,240 mile equatorial elliptical orbit. The total weight of the medium capability satellite including adapter and

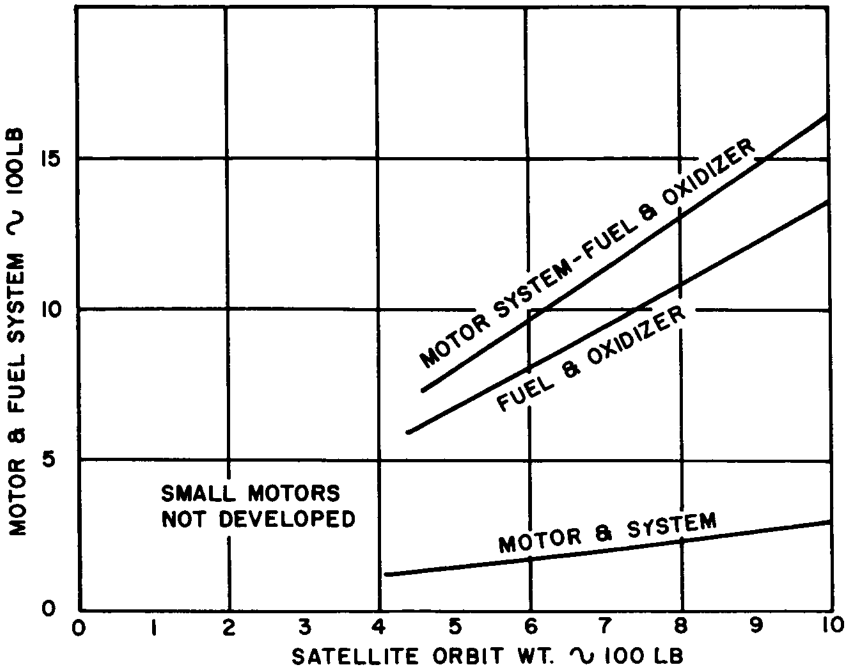


Figure 2-17. Liquid Apogee Motor Weight

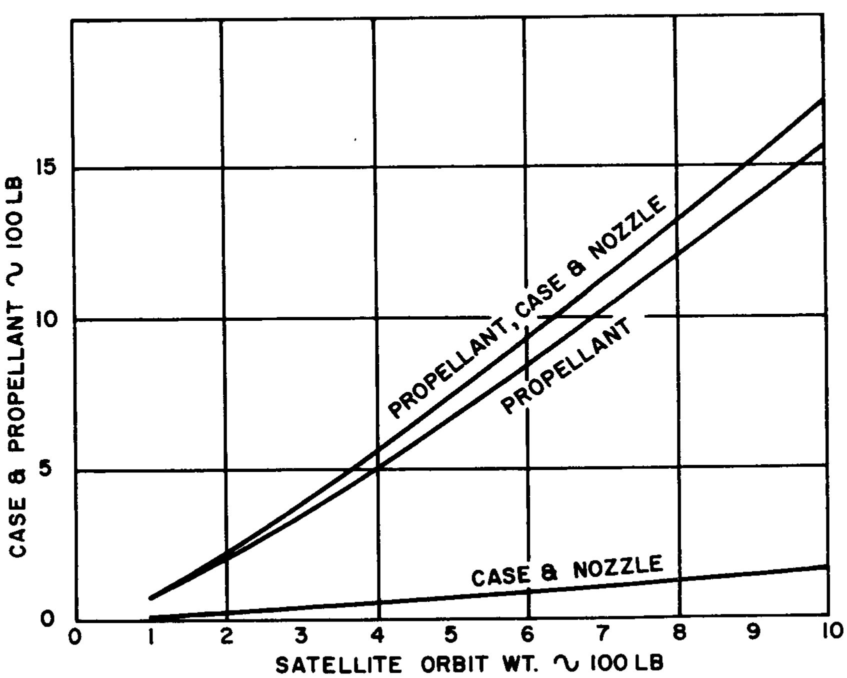


Figure 2-18. Solid Apogee Motor Weight

apogee motor is estimated at 1311 lb (see Table 2-10). The excess payload capacity of the booster provides reasonable margin for contingency and growth in the satellite.

The method of spin stabilizing the satellite and the location of the apogee motor during the ascent trajectory requires that the Agena reverse itself after final firing, spin up the payload, and then separate. The attitude control system and supply of control gas aboard the Agena is capable of this task.

9. Problem Areas

The design study has pointed out several problems that must be solved before a successful meteorological satellite can be placed in service. The problems can be defined as described below.

a. Sensors

An automatic Sun sensing system must be devised that will prevent direct impingement of Sun rays on sensitive receiving surfaces of vidicon tubes, image orthicon tubes, or infrared detectors. Direct Sun rays will instantaneously damage these devices beyond further usefulness.

Image orthicon and vidicon camera systems must be developed that will have a life of at least one year of unattended operation in space environment. Available equipment data indicates this service life does not presently exist. An auxiliary problem is that of developing a mirror and mechanism for the large number of cycles required by the mission life. All of these devices must be able to survive the severe stresses of launch and orbit injection.

b. Control

The attitude control system must be developed to a high degree of reliability and accuracy to achieve the platform stability required for sensors and lifespan of a year. A suitable Earth or horizon sensor must be developed to allow the stabilization system to work within the prescribed oscillation rates.

c. Structure

Structurally, the spacecraft must withstand all of its environments without undue penalties of weight. The dynamic problem of the passive despin must be considered. Passive thermal control means must be proved out, especially the heat of fusion materials proposed. It would be desirable to pursue the developments outlined in Appendix A. All of them offer significant weight savings and promise potential growth for the satellite.

SECTION 3 - HIGH CAPABILITY SPACECRAFT

A. MISSION OBJECTIVES

The design of the high performance vehicle was approached from two different aspects: 1) utilizing moderate sensor capabilities but achieving high reliability and long life through discreet use of redundancy, substantial margins on power and structure, and more conservative operational levels for equipment, and 2) seeking higher quality of information through the use of more sophisticated sensors. Both approaches assumed a satellite with an upper weight limit of 1000 lb. The Atlas-Centaur is proposed as the launch vehicle.

The first approach, using redundant equipment, must be carefully executed to prevent the systems, and even the redundant components themselves, from reducing the system reliability. This situation can result from failure in switching circuits, or faults in the mechanisms which detect and report failures. Generally, the first priority for duplication was given to components or systems whose failure would terminate the mission. In this category, loss of control gas in the attitude control system would be an example, as would failure of a vidicon or image orthicon tube. Systems whose failure would degrade the mission received secondary consideration. Relay receivers and telemetry are examples. Batteries also fall into this class; although peak powers, or transmission rates would be affected, the mission could be continued. Additional margin would be incorporated in the battery supply and solar array, permitting a greater degree of degradation before the dependent systems would be rendered inoperative. Additional weight permits the increase in structural margins of safety.

The second approach, acquisition of higher quality data, lends itself to such a wide choice of configurations that valid comparisons would be difficult. In the field of infrared detectors alone, the possible use of active cooling systems for the detectors could lead to systems with features quite unlike anything described in this section. Therefore, a logical conservative development in the high resolution optics that will provide a significant improvement in both day and night pictures will be used as a basis for the discussion.

B. SPACECRAFT CONFIGURATION AND PERFORMANCE

The configuration selected is a 3-axis stabilized satellite, very similar to the medium capability version. The circular section has been modified to accept a larger apogee motor and to be compatible with the Centaur booster. Solar paddles are used and are also folded twice, even though this is not necessary with the larger Centaur nosecone. Vibration problems and centrifugal force imposed on the vehicle during ascent dictate that the paddles be securely fastened. Due to the increase in size of the vehicle and larger payload capacity of the proposed booster, there is a greater latitude for selection and placement of sensors. Figure 3-1 depicts the proposed configuration.

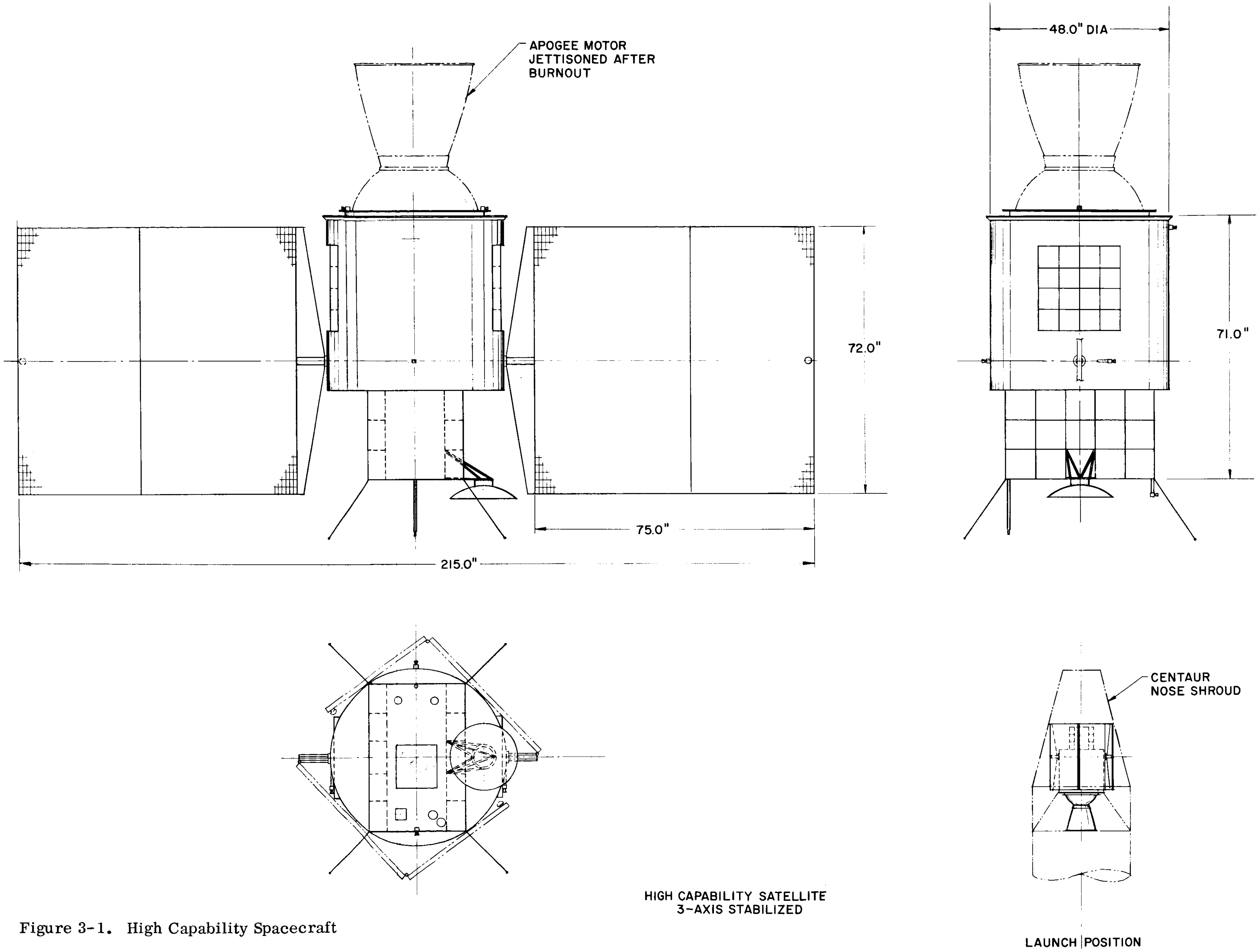


Figure 3-1. High Capability Spacecraft

Although the physical similarity of the medium capability and high capability satellites affords the opportunity for ready comparison between systems and equipment, it might be pointed out that the gravity gradient satellite may well lead to a more stable platform for sensors. The mechanical problems associated with gravity gradient decrease significantly as the size of the satellite approaches 1000 lb. A discussion of this type of satellite is presented in Section 5 of this volume.

Figure 2-3 is a subsystem block diagram applicable to this configuration. It shows the relationship of the systems that are described in subsection C.

Table 3-1 compares the capabilities of the two configurations. The only major difference is in the penetration of the high resolution optics. Characteristics common to both configurations are as follows:

- Communications Relay - Receive and retransmit data up to 100 KC base bandwidth on S-band (1700-2300 MC) using sensor data transmitter on time sharing or multi-frequency carrier operation
- Command Receiver and Telemetry - Operate on VHF using 148 MC and 136 MC, respectively
- Sensor Data and Communications Relay - Operate at S-band. Sensor data base bandwidth is 100 KC. The parabolic reflector antenna on the spacecraft is 21 in. in diameter and the ground antenna is 85 ft in diameter. Output S/N is 46 db with safety margin of 19 db. The communications relay uses the same spacecraft transmitter and antenna but assumes a 30 ft diameter ground antenna resulting in a 22 db S/N ratio with 7.5 db safety margin.
- Telemetry System - Uses PCM-FM with an information capacity of 200 bits/sec. The carrier to noise ratio (C/N) is 20 db.
- 3-Axis Control System - Maintains pointing accuracy to 1°. The system accuracy, basically capable of 0.1°, is limited by available horizon sensors. In normal operation, roll and pitch are controlled by variable speed reaction wheels acting on error signals from horizon sensors. In yaw (Earth pointing axis) a constant speed wheel performs the function of sensing errors and producing corrective torques. Cold gas jets are used for unloading the pitch and roll reaction wheels, and for station keeping. The spacecraft is maintained on station within ±2° in latitude and longitude.

Table 3-2 compares the weight of the two configurations and Table 3-3 gives a power supply summary. Although the weights differ considerably, the effect of higher resolution affects the power very little, requiring only the transmission of more data within a given time period. The average power supply has not been adjusted for the small amount, since considerable margin has been provided.

A comparison of these tables with Tables 2-1 and 2-2, showing weight and power of the medium capability satellite, will provide a rough estimation of the penalty paid for safety margin and reliability.

TABLE 3-1
CAPABILITY SUMMARY - HIGH CAPABILITY SATELLITE

	Medium Capability Sensors	Advanced Sensors
Full Earth Disc Picture		
Resolution at nadir*/Dynamic range	7 statue mi/daylight	7 statue mi/daylight
Shades of gray	8	8
High Resolution Picture		
Resolution at nadir/Dynamic range	1.3 statute mi-daylight to 3.3 statute mi at 1/2 Moon	0.7 statute mi-daylight to 1.3 statute mi starlight
Coverage per Frame	1250 x 1250 statute mi	700 x 700 statute mi
Frames per Cycle	25	49
Shades of Gray	8	8
Heat Budget		
Full Earth Coverage	200 statute mi res	200 statute mi res
Reflected Radiation	0.2 to 4.0 microns	0.2 to 4.0 microns
Emitted Radiation	4 to 40 microns in the range, 175 to 325°K	4 to 40 microns in the range, 175 to 325°K

* As used herein, the term nadir refers to the point on the Earth directly below the satellite.

TABLE 3-2
WEIGHT SUMMARY - HIGH CAPABILITY SATELLITE

Item	High Reliability Only (lb)	Improved Sensor Capability (lb)
Sensor Equipment	93.0	161.0
Communications	26.6	26.6
Data Handling	42.0	51.7
Power Supply	172.0	172.0
Attitude Control and Station Keeping (Subliming gas)	136.0	136.0
Wiring and Harness	26.5	35.5
Thermal Control	10.0	10.0
Structure	138.0	157.0
De-Spin Mechanism	34.9	24.9
Satellite Weight	669.0	774.7
Adapter	80.0	80.0
Propellant	890.0	998.0
Apogee Motor Case	100.0	112.0
Total Weight at Lift-Off	1739.0	1964.7

TABLE 3-3
POWER SYSTEM SUMMARY - HIGH CAPABILITY SATELLITE

Primary Battery	0 lb
Secondary Battery	67 lb
Solar Array	90 lb (75 ft^{2*})
Regulator-Selector	12 lb
Solar Paddle Drive	3 lb
Total	172 lb

* Required Solar Cell Area = 48 ft^2

System Requirements	Acquisition (W-hr)	Track Average (W)	Track Peak (W)
Attitude Control	79 W x 1 hr	105	241
Communications	60 W x 6 hr	55	83
Data Handling	31.1 W x 6 hr	36.4	36.4
Power Supply	7 W x 6 hr	7	7
Sensor Equipments	10 W x 6 hr	85	144
	727.6 W	288.4	511.4

One of the primary objectives for investigating the high capability satellite configuration was to determine what inherent reliability improvements would be achieved. A quantitative evaluation of reliability improvement was not made, because the specific system design had not been formulated. However, an estimate was made of the reliability improvements which would result from providing certain redundancies in the overall system.

For example, if the probability of a single junctional element failing during the mission is represented by Q_1, and the probability of its success by R_1, then

$$R_1 + Q_1 = 1, \text{ which corresponds to a certainty.}$$

Thus, for a single element, its reliability

$$R_1 = 1 - Q_1$$

If a duplicate element were incorporated into the system in a manner such that either the original or the added elements could perform the required function if either of them failed, then it would be necessary for both to fail in order to have a loss of functional capability. If the probability of failure is equal for both elements, and is independent of what happens to either element, then the probability of both failing, Q_2, is given by

$$Q_2 = Q_1^2$$

Accordingly

$$R_2 + Q_2 = 1 \text{ and}$$

$$R_2 = 1 - Q_1^2$$

Since Q_1 is a number less than unity, Q_1^2 is much less than Q_1 and therefore R_2 is greater than R_1.

To get a feeling for the numerical effect of a singly redundant element, assume that the probability of failure, Q_1, is 0.1. This would make a system with a single element 0.9 reliable. By adding a redundant element, the system reliability becomes $R_2 = 1 - (0.1)^2 = 0.99$.

For a similar system where the probability of failure was equal to the probability of success, 0.5 ($R, = 0.5$, $Q_1 = 0.5$), the addition of a single element results in $R_2 = 1 - (0.5)^2 = 0.75$.

This procedure may be extended to the case where three elements are provided to perform the function of any one of them. Then R_3, the combined reliability of the redundant system, is

$$R_3 = 1 - Q_1^3 .$$

The following table and Figure 3-2 illustrate the extent to which reliability can be improved through redundancy.

TABLE 3-4

Q_1	R_1	R_2	R_3
0.01	0.99	0.9999	0.999999
0.10	0.90	0.99	0.999
0.20	0.80	0.96	0.992
0.30	0.70	0.91	0.973
0.40	0.60	0.84	0.936
0.50	0.50	0.75	0.875

The following redundancies have been contemplated for the high capability satellite.

- Attitude Control - a complete duplicate system
- High Resolution Camera - duplicate vidicon tube (standby) and electronics
- Duplicate full Earth disc camera
- Telemetry Data Transmitter - a complete duplicate system
- Sensor Data Transmitter - a complete duplicate system

It is planned that the following systems will have increased operational life and/or reserve performance margin.

- Solar Array - 55% power margin
- Batteries - 25% power margin
- Structure - Structural margins obtained through a 3% increase in weight allowance (approximately 23% total orbital weight)

C. SUBSYSTEMS DESCRIPTION

1. Meteorological Sensors

The maximum capability satellite meteorological sensors will provide a higher degree of reliability through the employment of redundant backup systems. In addition, improved system performance will be afforded for the narrow coverage cloud cover and heat budget sensors by virtue of larger satellite volumetric, weight, and power abilities. The performance and physical characteristics of each meteorological sensor system are presented in Tables 3-5 through 3-7 as follows:

- Wide Coverage (daytime) Cloud Cover Sensor System - Table 3-5.
- Narrow Coverage (day-night) Cloud Cover Sensor System - Table 3-6.
- Heat Budget Measurement Sensor System - Table 3-7.

a. Wide Coverage Cloud Cover Sensor System

This system will provide improved reliability through the employment of a redundant backup system of identical form and characteristics as the prime sensor system. This sensor is also identical to that employed for the medium capability satellite. The performance characteristics of this system are summarized in Table 3-5.

TABLE 3-5
PERFORMANCE CHARACTERISTICS - WIDE COVERAGE CLOUD COVER SENSOR SYSTEM

Item	Characteristics
Sensor	1 in. Vidicon
Spectral Response	Visible region
Area Coverage	Full Earth disc
Resolution	7 statute mi at nadir per TV line
Number of TV Lines	800 minimum
Number of Gray Scales	8 steps of $\sqrt{2}$ difference
Operational Mode	Daytime surveillance
Operational Cycle	1 picture/30 min
Dynamic Range	35 to 1 minimum/1 frame
Automatic Exposure Control	Filter wheel and iris
Automatic Sun Protection	Capping shutter
Satellite Stability Requirement	0.003°/sec
Volume	375 cu in.
Weight	10 lb
Power, Peak	15 W

b. Narrow Coverage Cloud Cover Sensor System

This system will provide improvement in both reliability and performance. This is accomplished by using longer focal length optics and backup redundancy of two sensor systems which are identical to that configured for the medium capability satellite. The resolution will be increased from 1.3 mi to 0.7 mi at the nadir but at a 2 to 1 reduction of area coverage. Within the constraints of the sensor data handling system, the number of possible pictures will be increased from 25 to 49, thereby still providing a reasonable mosaic of a large portion of the Earth. Table 3-6 lists the performance characteristics of this system.

TABLE 3-6
PERFORMANCE CHARACTERISTICS - NARROW COVERAGE CLOUD COVER SENSOR SYSTEM

Item	Characteristics
Sensor	Miniaturized image orthicon
Spectral Response	Visible region
Area Coverage	625 x 625 mi
Resolution	0.7 statute mi at nadir per TV line
Number of TV Lines	1000
Number of Gray Scales	8 steps of $\sqrt{2}$ difference
Operational Mode	Daylight to 1/2 Moon surveillance
Operational Cycle	Up to 49 pictures/30 min
Dynamic Range	35 to 1 minimum/1 frame
Automatic Exposure Control	Filter wheel and internal tube functions
Automatic Sun Protection	Capping shutter
Satellite Stability Requirement	0.0015°/sec
Optics	f/2.0, 28 in. focal length, reflective lens

c. Heat Budget Measurement Sensor System

This system also provides for redundant backup and improved spatial resolution. The performance characteristics of this system are summarized below in Table 3-7.

TABLE 3-7
PERFORMANCE CHARACTERISTICS - HEAT BUDGET MEASUREMENT SENSOR SYSTEM

Item	Characteristics
Sensor	Thermister bolometer
Spectral Response	Visible and infrared
Area Coverage	100 x 8000 mi/scan
Temperature Resolution	1.0° minimum
Number of scans for full Earth	80 minimum
Operational Mode	Day and night surveillance
Operational Cycle	1 full Earth coverage/30 min
Operational time/cycle	2 min minimum
Automatic Sun Protection	Capping shutter
Satellite Stability Requirement	0.0015°/sec
Dynamic Range	175 to 325°K
Absolute Temperature Reference	Internal black body ± 2°K
Optics	f/1.5, 5 in. focal length reflective lens

2. Communications

The on-board communications subsystem of the high capability spacecraft will provide the following facilities:

(1) VHF command data receiver

(2) VHF telemetry data transmitter

(3) S-band meteorological sensor data transmitter

(4) S-band relay data receiver for reception and retransmission through a self-contained RF coherent translater driver channel utilizing the sensor data wideband output TWT stage

(5) S-band range and range tracking data retransmission to the ground utilizing the same equipment as in (4) above

(6) A multi-element S-band antenna system using a nondirectional slotted dipole array during the ascent phase and a directional parabolic antenna during the on-station phase. These antennas will be shared by the S-band transmitter and receiver functions by means of a duplexer.

(7) A common VHF omnidirectional antenna array for use by the command receiver and telemetry transmitter

A block diagram for the communications subsystem of the high capability SMS is given in Figure 3-2.

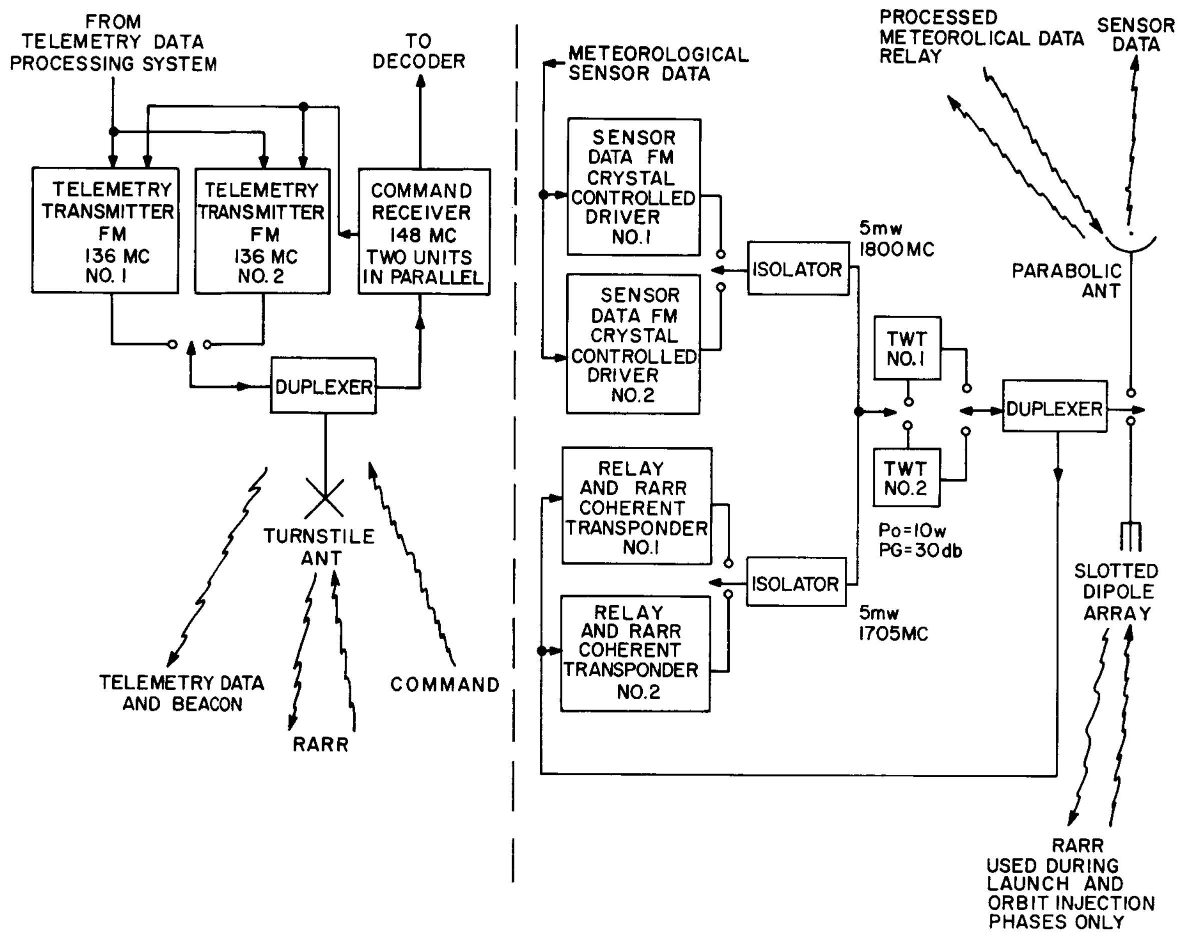


Figure 3-2. Communication System

The performance characteristics of the VHF and S-band portions of this subsystem are identical to those described for the medium capability SMS, previously described in subsection 2.D.2.

As shown in Figure 3-2, the only difference between the two subsystems is the provision for complete redundancy of all electronic S-band and VHF receiver and transmitter equipments. Switching facilities will be provided to permit the selection of any standby unit on command from the ground. The only exception to this will be the command receiver sections, which are essentially connected in parallel. Table 3-8 below lists the weight, volume, and power drain of the major units comprising the communications subsystem.

TABLE 3-8
SUMMARY OF CHARACTERISTICS - COMMUNICATIONS SUBSYSTEM

Quantity	Item	Total Weight (lb)	Total Volume (cu in.)	Peak DC Power Input (W)
2	Command Receiver	1.25	20	1.0
2	Telemetry Transmitter	2.8	46	10.0
2	Relay Transponder	6.0	200	5.0
2	Sensor Data Driver	4.0	150	5.0
2	S-Band TWT Output Stage	3.5	100	
2	TWT Power Supply	12.0	400	50.0
1	Turnstile Antenna	0.3		
1	Slotted Dipole Array	0.5		
1	Parabolic Antenna	3.5		
	Isolators, Duplexers, Switches	4.0		
	Total	37.85	916	71

3. Data Handling

The communications system aboard the spacecraft will permit the transmission of commands to the SMS, transmission of telemetry data to the ground, and retransmission of relay data and range rate data from the ground to the satellite and back to Earth.

a. Sensor Data

The data handling for the sensor in the high capability satellite is governed by the same requirements as those for the medium capability satellite. Although the high resolution picture has a somewhat greater information content (490,000 picture elements instead of 390,000) for the high capability satellite, the requirement for a full Earth disc picture is the same and exceeds the high resolution needs. The full Earth disc picture has 640,000 picture elements.

b. Command Data

The data handling portion of the command link is essentially the same for the high capability satellite as for the medium capability satellite. Estimates for command requirements on the part of the various systems designers indicate a very slight increase for controlling redundant systems. This will not affect the choice of the command system, since, as presently designed, it is capable of expansion to a total of 256 commands.

c. Telemetry Data

Monitoring of additional points is needed in the communications section to effectively utilize the redundant equipment added for increased reliability. However, the items to be checked are all in the once per minute range, so that the total increase amounts to less than two pulses per second or an insignificant amount in terms of the design of the data handling section of the telemetry.

4. Attitude Control

The high capability satellite has a 3-axis stabilization system which is similar to that of the medium capability satellite but offers increased reliability obtained through the use of redundancy. The control system stabilization rates and pointing accuracy are the same as the medium capability satellite. Figure 2-8 illustrates the proposed system.

An approach which has sufficient merit to warrant further study is the attainment of redundancy through use of dissimilar equipment performing the same function as the basic equipment. This can increase both reliability and accuracy. Some possibilities of this approach are described below.

The addition of a laser attitude sensor system, as described in Section 2 of Volume 4, will greatly improve the accuracy of the control system. Since this system performs the same function as the horizon sensor, the two systems are redundant.

As an alternate approach, the addition of a Polaris tracker and a single gimballed fine Sun tracker can provide greater attitude measurement accuracy than the medium capability system. Although these sensors can measure star and Sun lines to seconds of arc accuracy, computation is necessary to convert these angles into accurate attitude angles. The overall attitude sensing system including computation will have an accuracy of better than a minute of arc. Use of the turn gyro or GLOPAC torquing systems can provide control system stabilization rates of 10^{-4}°/sec and a pointing accuracy of a few minutes of arc.

The additional weight allowance will permit the use of a derived rate type of control gas system. This system will have better performance during acquisition than the regular cold gas system, and it can also be used for attitude control in the tracking mode instead of the gyro torquing systems.

Discussion of these systems can be found in Volume 4.

5. Power Supply

Solar cells and nickle cadmium batteries are used for the power supply system on the high capability satellite. While the effect of redundancy has had a substantial effect on the weight, it has had little effect on power supply, as most of the added equipment is on a complete standby status, or uses only nominal power to maintain proper temperature.

One basic change in the system is the elimination of the "one shot" primary batteries. Instead, secondary batteries with more than adequate power for acquisition and track requirements are supplied. These batteries are capable of sustaining full system operation during occult periods. Because of their high capacity, simultaneous operation of all systems is possible. Depth of discharge during normal operation is sharply reduced resulting in an increase in reliability. Table 3-3 summarized the estimated power requirements of the proposed configurations.

6. Thermal Control

Thermal control methods on the high capability configurations utilize the methods described for the medium capability satellite; namely, shaded oriented panels, insulation, heat sinks, and local heaters. Table 3-9 summarizes the heat loads and temperature limits of the equipment.

7. Structure

The satellite consists of a hollow circular section surmounted by a box-like section housing the sensors and relay equipment. The cylinder houses the telemetry, power supply, attitude and station keeping system, and the majority of the communications equipment, as well as the apogee motor. The lower end of the cylinder is terminated by a structural frame supporting the de-spin weights and deploying mechanism. This frame also contains provisions for attaching the satellite to the booster. The proposed means of separating the satellite from the booster are to use either a Marman type clamp released by explosive bolts, or to cut the adapter with a shaped explosive charge. The upper end of the cylinder terminates in a structural bulkhead upon which a variety of sensor and communication equipment may be mounted. The apogee motor is jettisoned. The jettison takes place after the satellite is de-spun and the solar panels extended and rotated 90° to allow the clamps and debris to have ample clearance.

In the cylindrical section, axial loads are taken mainly by four full length longerons and distributed to the skins. Intermediate shelves and baffles serve as auxiliary load carrying members for equipment. A section of the outer skin in replaced by flat panels which are oriented toward space when the spacecraft is in orbit. The majority of equipment which must be thermally controlled is mounted on these panels. The panels are protected from incident sun rays by a hollow grid "shadow box." The combination of space-oriented panels and proper coatings permits temperatures to be controlled within the required range (25±10°C). A similar treatment is extended to sensor equipment. Equipment located in other areas is thermally controlled by insulation, heat sinks, heat-of-fusion material and local heaters.

8. Propulsion

a. Apogee Motor

Solid propellant apogee motors were sized for both high performance satellites. Studies of the relative merits of solid propellant motors versus liquid propellant motors, which resolved in favor of the solid, were completely valid for the weight class of this satellite. A Δv of 6000 ft/sec was

TABLE 3-9
HEAT LOAD SUMMARY HIGH CAPABILITY SATELLITE

Upper Shadow Box					
Component	Heat Dissipation (W)	Temperature Limits	Component	Heat Dissipation (W)	Temperature Limits
Battery	25	25±10°C	Battery	25	25±10°C
Battery	25	25±10°C	Battery	25	25±10°C
Attitude Control	28	25±10°C	Telemetry	9.5	25±10°C
Command Receiver	12	25±10°C	Telemetry	9.5	25±10°C
Data Programmer	2	25±10°C	Relay Receiver	3	25±10°C
Telemetry Command	1	25±10°C	Telemetry Command	1	25±10°C
Total	93		Total	73	

Lower Shadow Box					
Component	Heat Dissipation (W)	Temperature Limits	Component	Heat Dissipation, (W)	Temperature Limits
Sensor Data Transmitter	92	25±10°C	Relay Receivers	6.5	25±10°C
High-resolution Camera	29	40±5°C	High-resolution Camera	25	40±5°C
Heat Budget Sensor	60	25±10°C	Low-resolution Camera	11	25±10°C
Horizon Scanner	4	25±10°C	Horizon Scanner	4	25±10°C
Total	185		Total	46.5	

Heat Storage Material Controlled Components		
Component	Heat Dissipation (W)	Temperature Limits
6 Reaction Wheels	30	25 ± 10°C
6 Gyros	10	25 ± 10°C
2 Regulators and Inverters	7.2	25 ± 10°C
Total	47.2	

assumed for the apogee kick. Figure 2-18 was used to estimate the motor weight which is shown in the weight summary, Table 3-2.

b. Launch Vehicle

The Atlas-Centaur, the proposed boost vehicle, has ample capability to lift either of the proposed high performance satellites into synchronous orbit. Although the precise capability of this booster is not known, the estimated weight of a satellite that can be put in a 22,240 nm equatorial elliptical orbit exceeds 1000 lb, leaving a contingency and growth potential of over 200 lb for the satellite.

The Atlas-Centaur is composed of two stages, a first stage liquid propellant Atlas D, having a rated sea level thrust of 367,000 lb, and a second stage Centaur, also using liquid propellant, having a capability of more than one restart and thrust of approximately 30,000 lb. Its guidance system is capable of the 180° turn around specified for the SMS ascent and will probably have more precise sensors and control system, resulting in injection errors lower than those expected of Agena B.

D. PROBLEM AREAS

Since the study made for the high capability satellite results in a configuration that is similar, if not identical to that of the medium capability configuration, most of the problem areas are the same. The new problems have to do with the complications caused by redundancy and duplication. In order, they are:

(1) Means of sensing failure or sufficient degradation of the vidicon sensor, command channels, other redundant electronics. Physical means to divert the optical path to the standby image orthicon tube. The standby tube must endure the stresses of launch and then be kept in a tolerable environment until it is switched into the system. The switching circuit and mechanisms are assumed quiescent until required.

(2) The standby attitude control system requires the same considerations as the vidicon tube. To be useful when necessary, it must endure some time in space environment without deterioration. Valves, nozzles and other working parts must not cold weld. Gas leakage must not occur.

In addition, the following problems must be solved. They are most important, since the life of the satellite may well exceed one year.

(1) Sun sensing and screening for delicate detectors

(2) Reliable operation of unattended vidicon and image orthicon tubes.

(3) Reliable mechanism for the optical scanning systems.

(4) Reliable horizon or Earth sensors must be developed.

(5) Reliable rate gyros must be developed.

SECTION 4 - MINIMUM CAPABILITY SPACECRAFT

A. MISSION OBJECTIVE

A study was made to determine the quality of weather information that could be obtained with a small satellite weighing not more than 100 lb. The economic advantages of a satellite that could be orbited by a relatively low cost launch vehicle are obvious. From the study it was concluded that the minimum return of information that would justify the effort to construct and orbit the satellite would include:

(1) Full Earth disc picture taken under daylight conditions

(2) Heat budget measurements of emitted and reflected radiations

(3) One complete set of unprocessed data transmitted every 30 minutes

(4) Capability of relaying selected processed data to Nimbus and other ground stations

The Thor-Delta would be the launch vehicle for this mission. Its physical characteristics were used as a basis for part of the study.

B. CONFIGURATION

A spin stabilized satellite was chosen as the most likely configuration to accomplish the mission. When finally oriented, the spin axis is normal to the orbit and equatorial planes, roughly parallel to the Earth axis. This configuration, shown in Figure 4-1, offers the following attractive features:

(1) Adequate stabilization, using the vehicle's own motion for maintaining orientation

(2) Simple and adequate thermal control

(3) Constant power level from the solar array, due to orientation of the spin axis

(4) Reduced despin requirement

The major problem associated with a spinning satellite is the necessity for image motion compensation. Complexity and weight are introduced by the systems required to despin the Earth image to allow proper light exposure and scanning.

Table 4-1 compares the weight of the three types of satellites, spinning, gravity gradient and 3-axis stabilized, for the same mission and equipment. It can be seen that the weight penalties for stabilization and structure make the gravity gradient and 3-axis stabilized configuration less attractive, even though they provide the better sensor platforms.

The factors detracting from the performance of a spin stabilized satellite as a sensor platform are a deviation of the axis caused by solar flux (basically

TABLE 4-1
WEIGHT COMPARISONS
MINIMUM CAPABILITY SATELLITE

Item	Spin	3-Axis	Gravity Gradient
Sensor Equipment	20.0	20.0	20.0
Communications	14.4	14.4	14.4
Data Handling	19.5	19.5	19.5
Power Supply Total			
a. Body Cells	60.3	90.2	a. .136.4
b. Oriented Paddles			b. 108.9
c. Fixed Paddles			c. 117.4
Batteries	26.7	43.2	42.9
a. Primary	8.5	9.7	9.5
b. Secondary	6.2	21.5	21.4
c. Charger	12.0	12.0	12.0
Body Mounted Solar Cells	33.6		a. 93.5
Oriented Solar Paddles		47.0(1)	b. 66.0(2)
Fixed Solar Paddles			c. 74.5(4)
Attitude Control and Station Keeping	26.6	40.4	43.5
Wiring and Harness	13.5	16.0	16.0
Separation Penalty (64 ft) (Connecting Circuits)			10
Thermal Control			
Body Mounted Solar Cells			a. 15.0
Structure	48.0	59.0	67.0
Body Break for Separation			a. 5.0
Interconnect Beams			b. 11.0
Despin Mechanism	16.0	18.8	18.8
Satellite Weight	218.3	278.3	
a. Body Cells			a. 360.6
b. Oriented Paddles (Solar Array)			b. 318.1
c. Fixed Paddles			c. 326.6

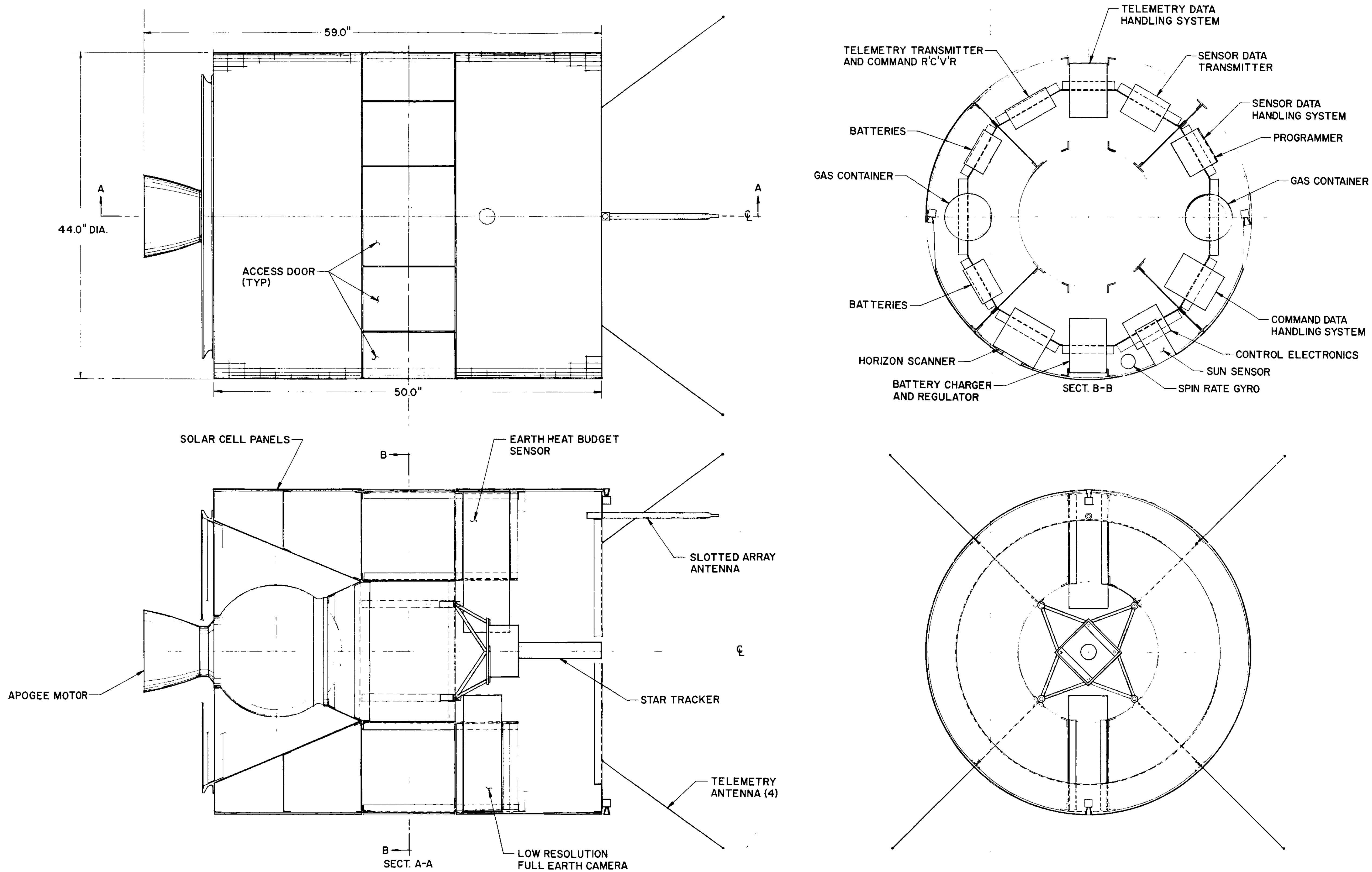


Figure 4-1. Minimum Capability Spacecraft

a sinusoidal force) and wobble caused by physical unbalance. Wobble can be caused either by imperfect original balancing that results in the spin axis deviating from the principal axis, or by damage occurring during launch or in space, such as loss of a solar cell module. Figures 4-2 to 4-4 indicate the effect of these disturbances.

Figure 4-5 shows the relationship of various subsystems contained in the spacecraft. These subsystems are described in detail in subsection D of this section.

C. SPACECRAFT PERFORMANCE

One of the prime considerations of a low capability satellite was the design of a system with the lowest possible weight, targeted at 100 lb. Detailed studies of the minimum acceptable sensors coupled with data transmission power requirements led to systems having total orbital weights above 200 lb. Table 4-1 summarizes the weights of various configurations that have been investigated in attempts to arrive at minimum system weights. Although reducing capability, accuracy, and life may lead to substantial reductions of weight, it is not felt that the resulting system will return sufficient meaningful data to be worthwhile.

The performance of the satellite is outlined here, with detailed descriptions of the subsystems following in subsection D.

The sensors will provide:

(1) Full Earth disc picture using a vidicon camera
Resolution at nadir* - 7 statute mi per TV line
Dynamic range - daylight conditions
Cycle time - one (1) picture every 30 min
Shades of gray - 8

(2) Heat budget measurements using an infrared detecting device
Full Earth coverage with 8000 mile resolution
Emitted radiation 4 to 40 microns in range of 175 to 325°K
Reflected radiation in the 0.2 to 4 micron range

(3) Communication relay
Will receive and transmit data up to 1.5 KC baseband width; slow speed, facsimile, or teletype RF carrier (136 MC) using the telemetry transmitter on a time shared basis.

The spin stabilized control system will maintain the alignment of the spin axis within 1° of the desired orientation. A Polaris star tracker will be used as the sensor. Since the actual position of the star Polaris is approximately 55 minutes off the normal to the equatorial plane, the satellite spin axis will be

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* As used herein, the term nadir refers to the point on the Earth directly below the satellite.

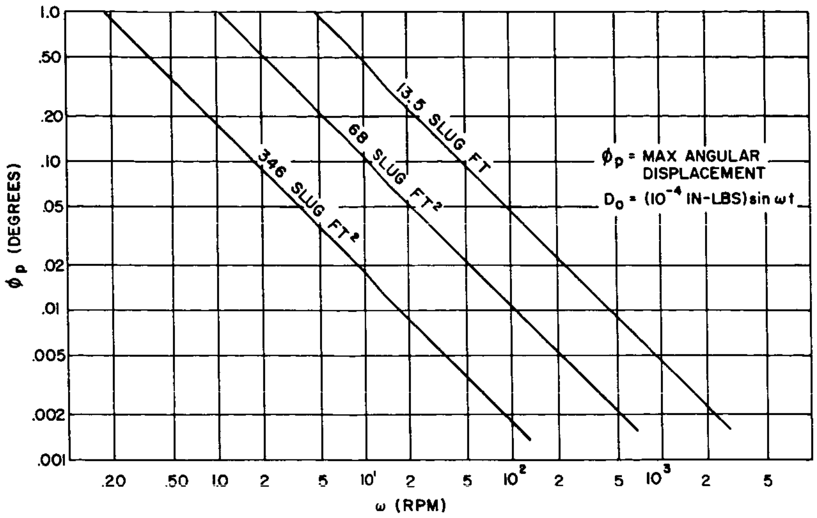


Figure 4-2. Angular Displacement vs Spin Rate

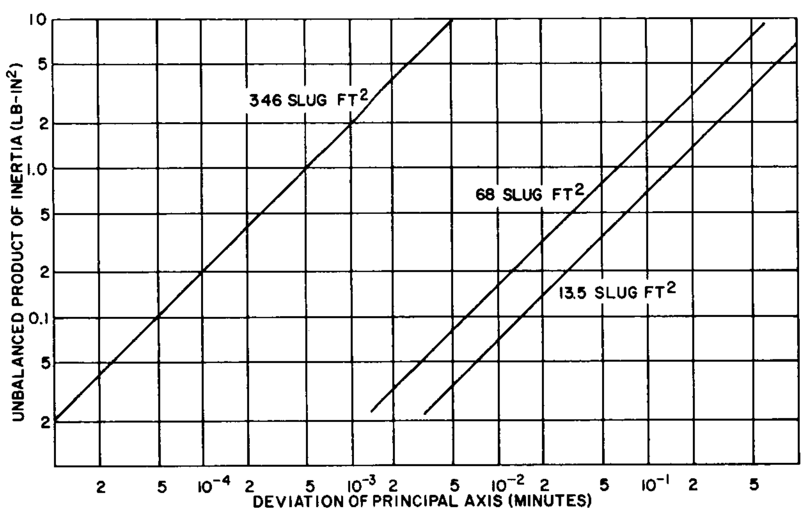


Figure 4-3. Deviation of Principal Axis Due to Unbalance

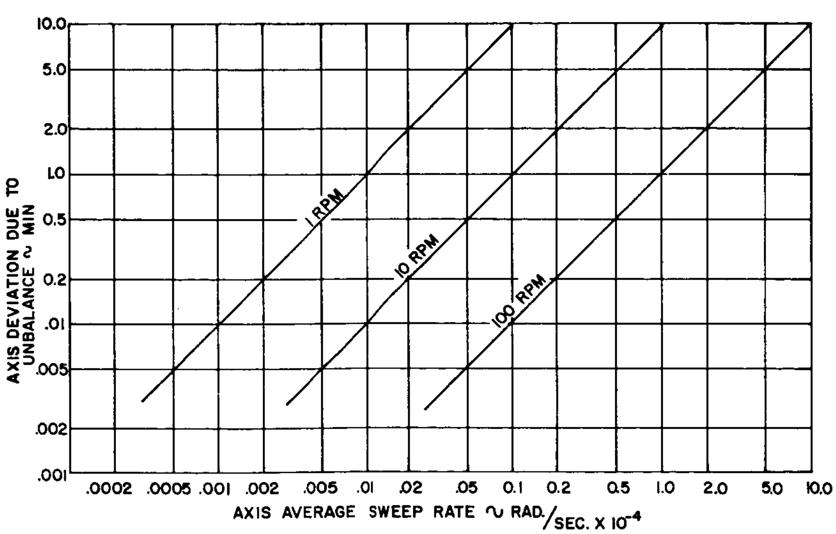


Figure 4-4. Principal Axis Motion Due to Unbalance

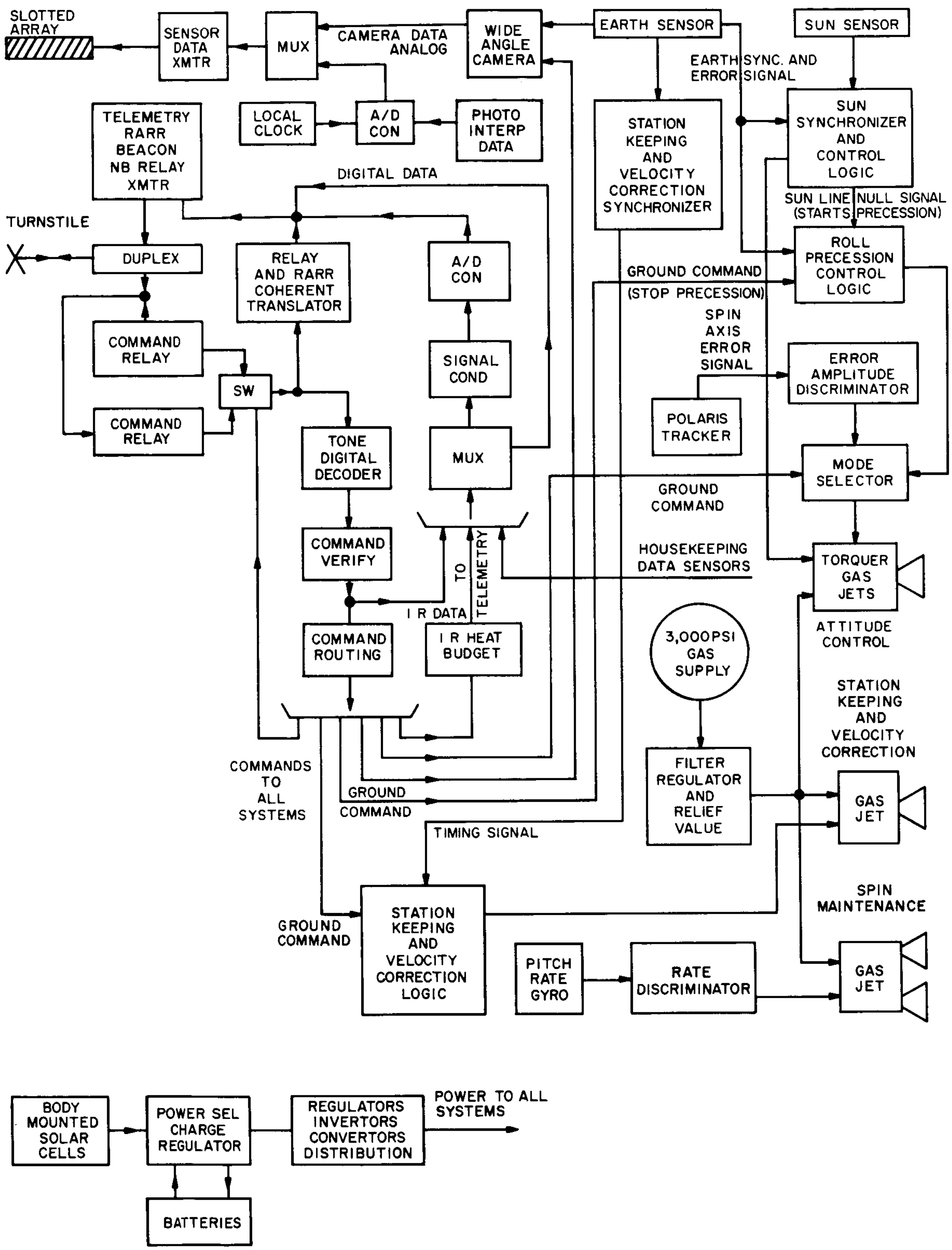


Figure 4-5. Minimum Capability Spacecraft Subsystems

rotated by this nominal amount. The cameras and heat budget sensors are located radially in the satellite normal to the spin axis. A horizon sensor and Sun sensor will be used as timing devices for shutter operating sequences. Both attitude and station keeping are accomplished by pulsed cold gas jets. The spacecraft will be maintained on station within ±2° in latitude and longitude.

The overall system is shown in the block diagram Figure 4-5. The command receiver, relay and telemetry operate on VHF, the command receiver at 148 MC and the relay and telemetry at 136 MC. Sensor data operates on S-band.

D. DESCRIPTION OF SUBSYSTEMS

The various subsystems in the spacecraft, and the factors determining their choice and operation will be discussed in this subsection. Because maintaining minimum weight was a prime consideration in designing this configuration, many compromises were decided on this basis. Systems were based on known and tested techniques because the weight savings possible by use of untested methods would not have reduced the weight sufficiently to use the proposed launch vehicle.

1. Meteorological Sensors

The minimum capability satellite meteorological functions will be provided by a cloud cover surveillance sensor system and a heat budget measurements system. Constraints are imposed by the satellite stabilization system and weight, power, and volume to accomplish a limited scope of meteorological surveillance. The necessary sensor systems and their performance capabilities and characteristics are described in the following paragraphs.

a. Full Earth Coverage, Daytime, Cloud Cover Sensor System

The full Earth coverage cloud cover sensor system configured for the minimum satellite is the identical system used in the medium capability satellite. The only change required is the incorporation of a dual purpose stop motion and Sun protection shutter. The performance characteristics, volumetric, weight and power requirements are listed in Table 4-2.

b. Full Earth Coverage, Heat Budget Measurements Sensor System

The sensor system provided for measurement of the Earth's heat budget is shown by Figure 4-6. This sensor uses geometric cones and the satellite spin rate to accomplish scanning of the full Earth disc without the use of mirrors or lenses. Heat budget measurement of the entire Earth is provided as a single integrated data point. The performance of this system is summarized in Table 4-3.

TABLE 4-2
FULL EARTH COVERAGE CLOUD COVER
SENSOR SYSTEM PERFORMANCE

Item	Characteristic
Sensor	1 in. vidicon
Spectral Response	Visible region
Area Coverage	Full Earth disc
Resolution	7 statute mi at nadir per TV line
Number of TV Lines	800 minimum
Number of Gray Scales	8 steps of $\sqrt{2}$ difference
Operational Mode	Daytime surveillance
Operational Cycle	1 picture/5 min
Dynamic Range	35 to 1 minimum/1 frame
Automatic Exposure Control	Filter wheel
Automatic Sun Protector	Capping shutter
Shutter speed	0.0002 sec max
Satellite stability	10 RPM
Volume	375 cu in.
Weight	10 lb
Power, peak	15 W

TABLE 4-3
HEAT BUDGET MEASUREMENTS
SENSOR SYSTEM PERFORMANCE

Item	Characteristic
Sensor	Thermistor bolometers
Spectral Response	Visible and infrared
Area Coverage	Full Earth
Temperature Resolution	1.0°
Number of Scans for full Earth	1
Operational Mode	Day and night
Operational Cycle	1 full Earth coverage/30 minutes
Automatic Sun Protection	Capping shutter
Satellite stability requirement	10 RPM
Dynamic Range	175°K to 325°K

The heat budget sensor contains the following major components.

1) Detector Head Subassembly. This subassembly consists of the two geometric cones and the two detectors, one black and the other white.

2) Control Chassis Subassembly. This subassembly contains all the circuits and the power supply necessary to process the received radiant energy into a data form suitable for input into the sensor data handling system.

3) Sun Protection Shutter. A capping type shutter is provided to prevent the Sun's image from impinging on and damaging the system detectors. Shutter operation is initiated by a signal from a Sun sensor.

The volumetric, weight and power requirements are listed below:

Volume	125 cu in.
Weight	5 lb
Power, Peak	5 W

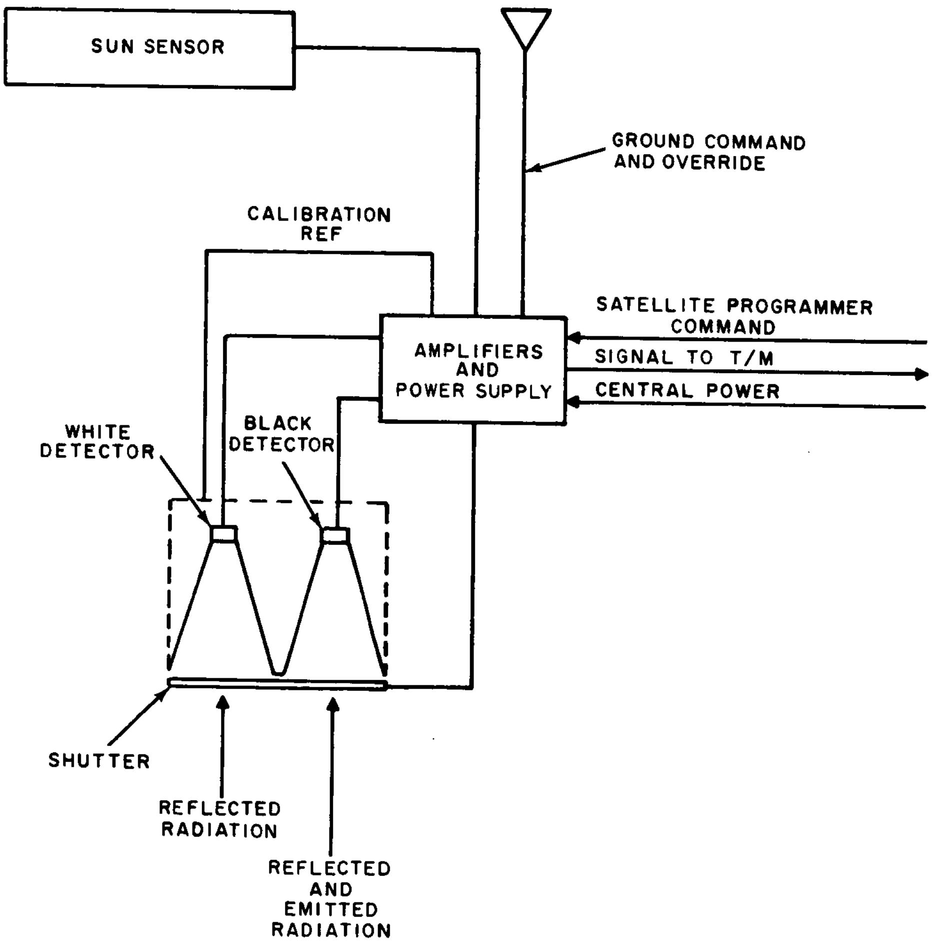


Figure 4-6. Wide Beam Heat Budget Sensor

2. Communications

The communications system aboard the spacecraft will permit transmission of commands to the SMS, transmission of telemetry data to the ground, and retransmission of relay data and range rate data from the ground to the satellite and back to Earth, and transmission of sensor data to the ground.

The communications subsystem on board the low capability spacecraft will provide the following facilities:

(1) S-band meteorological sensor data transmitter

(2) VHF command receiver

(3) VHF relay data and retransmission through an RF coherent translator driver channel using a portion of command receiver

(4) VHF range and range rate tracking data retransmission utilizing the same equipment as in (3) above

(5) VHF telemetry FM driver

(6) A common VHF output stage

(7) A common VHF omnidirectional antenna array for use by the command receiver and VHF transmission channels

(8) S-band slotted dipole antenna array with 6 db gain having an omnidirectional pancake beam

A block diagram of the entire system is shown in Figure 4-7. This system will permit time-share transmission of the telemetry, relay, or range and range rate data.

a. S-Band System

The S-band equipment proposed herein, for the low capability SMS, will permit the transmission of meteorological sensor data while the vehicle is on station. The equipment will have the following characteristics.

Characteristic	Ground Link
Base Bandwidth	100 KC
Output S/N	46 db
Ground Antenna Size	85 ft
Safety Margin	3 db

A direct frequency modulator FM generator will be used in the S-band transmitter (see Volume 5). A frequency multiplier consisting of two low power cascaded varactor quadrupler stages developing approximately 5 MW at 1800 MC will be used. This unit will feed the TWT directly. The significant characteristics of this equipment are as follows:

(1)	TWT Power Output	2.5 W
(2)	Base Bandwidth, Max.	100 KC
(3)	Modulation Index	16.3
(4)	Driver Size	5 x 5 x 3 in.
(5)	Driver Weight	2 lb
(6)	TWT Weight	1.5 lb
(7)	TWT Size	1-1/2 dia x 11 in.
(8)	High Voltage Power Supply Size	6 x 4 x 5 in.
(9)	High Voltage Power Supply Weight	4 lb
	Total Weight	7.5 lb

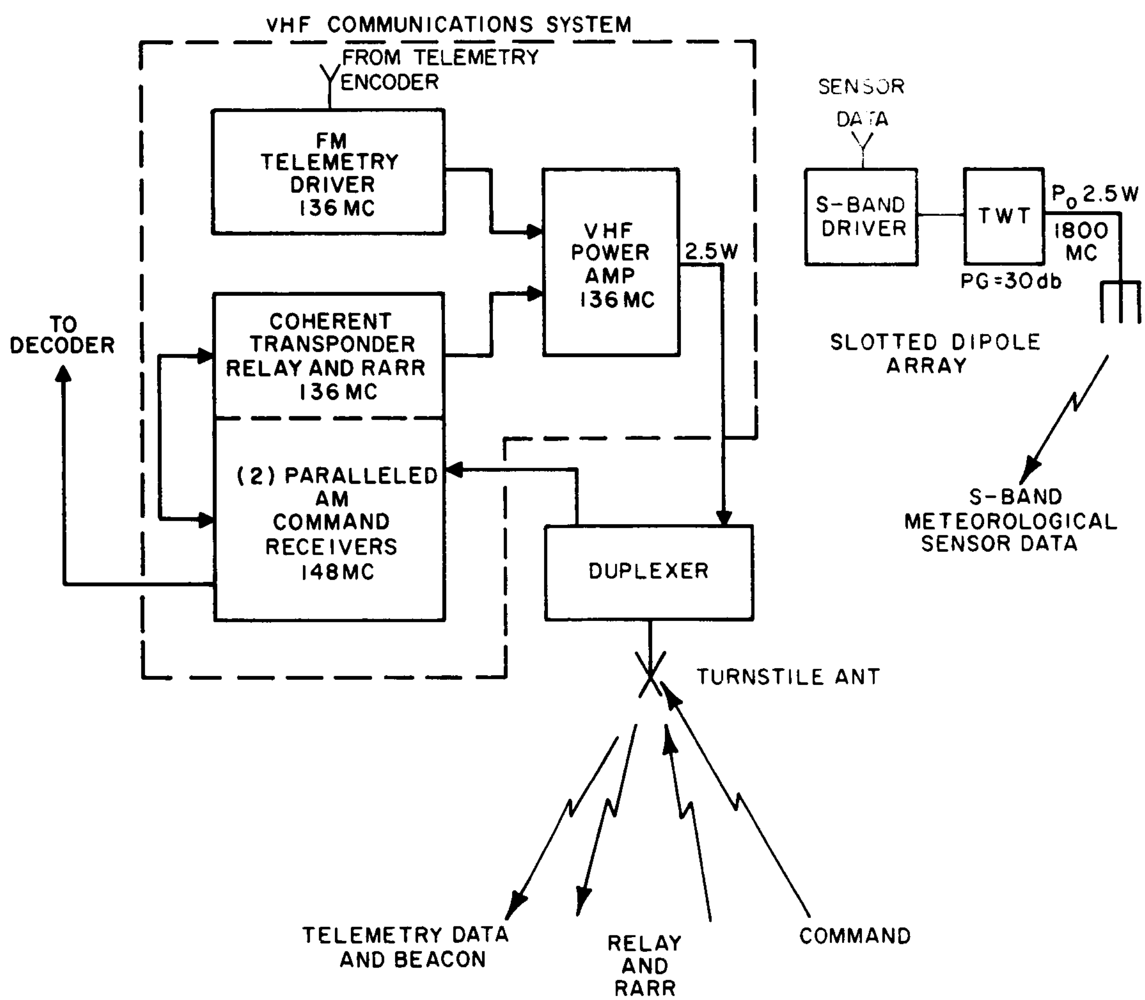


Figure 4-7. Communication System

The TWT output will be fed directly to the slotted dipole array mounted on the SMS exterior surface.

b. VHF System

Inspection of the block diagram Figure 2-8 reveals that the entire system is integrally related because this system must handle the communication relay retransmission and range and range rate tracking functions in addition to its normal command reception and telemetry transmission functions.

This is accomplished by making the command receiver an integral part of the coherent transponder driver used for both relay and range and range rate operation. Telemetry operation is accomplished by using a separate FM crystal controlled modulator driver. A common RF amplifier output stage is time shared by both of the inputs just described. A common VHF omnidirectional turnstile antenna is used by both the VHF receiver and transmitter channel.

The entire unit is enclosed in one package. No redundancy is provided for increased reliability except for the inclusion of two command receiver portions connected in parallel.

Since both units are interrelated, a common set of characteristics is presented.

Characteristics	
Command Receiver Frequency	148 MC
Command Receiver Sensitivity	1 (μv)
Command Receiver IF Bandwidth	60 KC
Command Receiver Video Bandwidth	20 KC
Command Receiver Image Rejection	60 db
Relay (RARR) Driver Retransmission Frequency	136 MC
Maximum Relay Modulation Frequency	1.5 KC
Maximum RARR Modulation Frequency	20 KC
Power Amplifier Output Power	2.5 W
VHF Unit Weight	4 lb
VHF Unit Volume	100 cu in.
DC Power Input	15 W

The VHF antenna will be a turnstile type having omnidirectional characteristics. It will consist of four spring mounted quarter wave whips (approximately 22 in. long) at 90° intervals around the satellite circumference. A suitable duplexer will provide adequate receiver isolation from the radiated transmitter signal.

c. Summary of Subsystem Characteristics

Listed in Table 4-4 are the weights, volumes and power drains of the units composing the communications subsystem.

TABLE 4-4
COMMUNICATIONS SYSTEMS POWER SUMMARY

Quantity	Item	Weight (lb)	Volume (cu in.)	DC Power Input (W)
1	VHF Receiver Transmitter	4	100	15
1	S-band TWT	1.5	20	
1	S-band Driver	2.0	75	5
1	High-voltage Power Supply	4.0	120	15
1	Turnstile Antenna	0.3		
1	Slotted Dipole Array	0.5		
	Totals	12.3	315	35

3. Data Handling

a. Command Data

An estimate of the number of commands needed in the minimum performance vehicle comes to just over eighty. Thus, a tone digital command system which presently handles 70 commands would require some modification. For the minimum capability vehicle, it is desirable to use only one command system. The existence of a network of tone digital stations makes that system attractive. A method of using this system for many commands is described in subsection 2.B.2 of Volume 5. It is likely, moreover, that as a minimum system develops, the number of commands actually needed will drop below seventy.

Such a system will occupy 154 cu in., weigh 6.9 lb, and require 9.8 W.

b. Telemetry Data

There is virtually no change in the telemetry data handling requirements in going from the medium performance to the minimum performance vehicle. A slight reduction in the number of points to be monitored has no effect on the system. The only point worth noting (from a system standpoint) is that the data from the heat budget sensor, in this vehicle, will be transmitted to the ground as part of the telemetry data rather than over the sensor link, as is required in the other vehicles.

c. Sensor Data

The data handling for the sensor in the minimum capability satellite is governed by the requirement for transmitting a daylight full Earth disc picture. Using a frame which is 8000 miles by 8000 miles, there are 640,000 picture elements to be transmitted per frame. There are no high resolution pictures taken by this vehicle. The requirement for transmitting heat budget information can be accommodated by the telemetry link, because the amount of information is extremely small.

It is possible to consider reading off the picture over a five minute period without losing it. Because only the Earth disc picture will be obtained, the system time schedule will allow for a five minute readout. If this is done, the total information rate is 640,000/300, or approximately 2.2 kilobits/sec.

At the same time, because the emphasis in the minimum performance vehicle is on stripping away features which are not absolutely necessary, the entire section involving the addition of photointerpreter data will be omitted. This has little effect on the data but does eliminate some hardware with its consequent reduction in power requirement.

4. Attitude Control

A spin stabilized vehicle will be used for the minimum capability satellite. The passive despin system for this vehicle will reduce the spin rate to approximately 10 RPM from its ascent trajectory spin rate. In its final orientation the spin axis will be inertially fixed and normal to the orbit plane. The control system is shown in Figure 4-8. After passive despin, the spin rate of the satellite is maintained at 10 RPM by a control system composed of a rate gyro, rate discriminator logic, and cold gas jets. The physical location of the gas jets is shown in Figure 4-9. Acquisition of the Sun is accomplished by using a Sun sensor with a fan shaped field of view in a plane which contains the spin axis. A null signal is obtained when the Sun line is normal to the spin axis. Control motion is obtained by applying a control torque impulse which causes the vehicle to precess about the initial spin axis position. A second control impulse is applied when the vehicle has precessed through 180° to stop the precession, producing a net displacement of the spin axis.

When null is achieved with the Sun line, the vehicle is caused to precess in roll; that is, the vehicle rolls about the Sun line. When the fan shaped Earth sensor indicates a null, the roll precession is stopped by ground command. Manual precession control is used because ground control must determine if the vehicle is in the proper orientation or upside down. The time sequence between the Sun and Earth sensor signals will provide the necessary information. Because this relationship continually changes (depending on the time of day) manual control is much simpler than automatic on board control.

After the Earth has been acquired and tracked for six hours, the spin axis of the vehicle will be normal to the orbit plane within the accuracy of the Earth sensor control. The Polaris boresight tracker will then be energized and the Earth

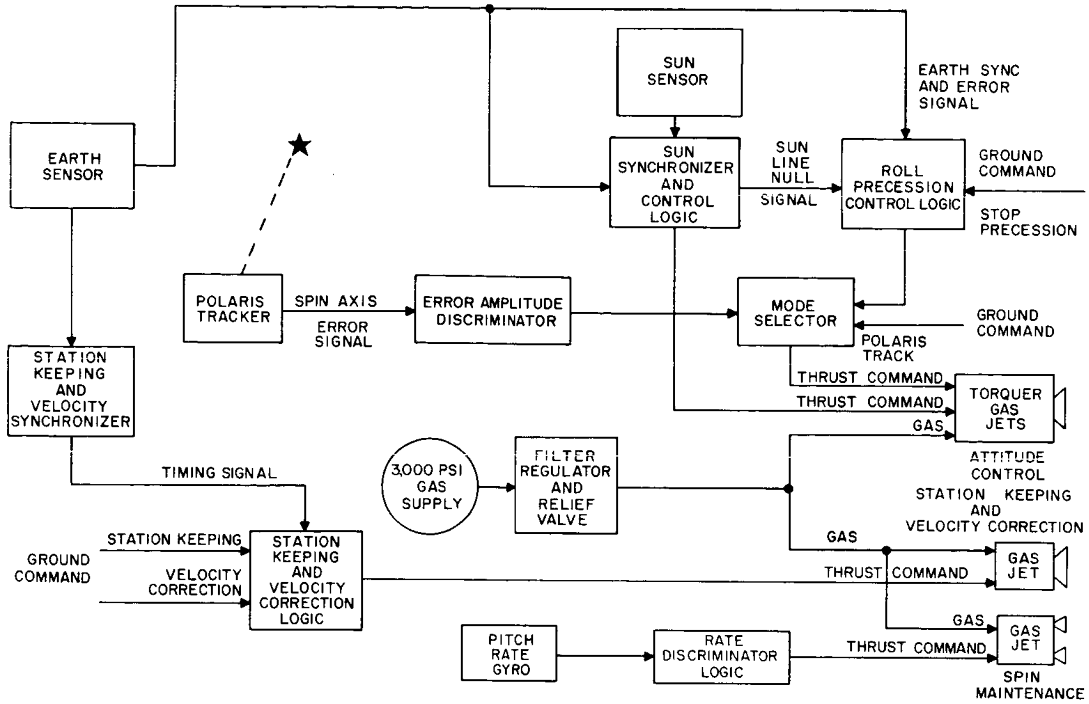


Figure 4-8. Spin Stabilized Control System

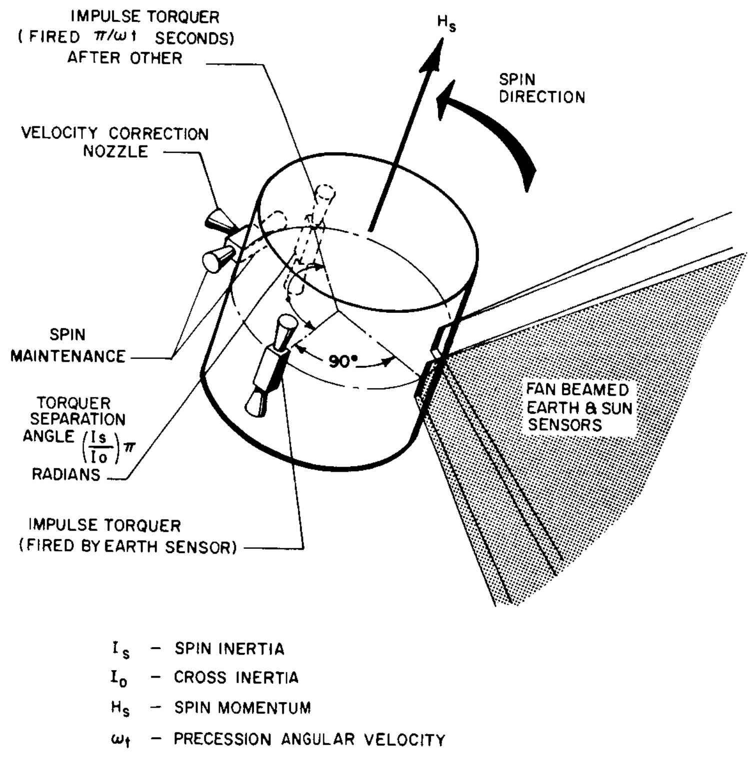


Figure 4-9. Location of Gas Jets for Spin Stabilized Vehicle

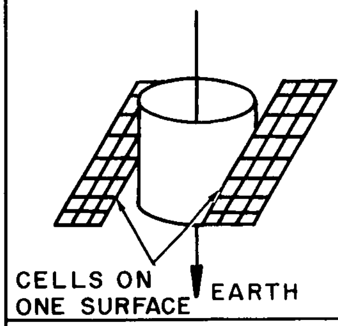
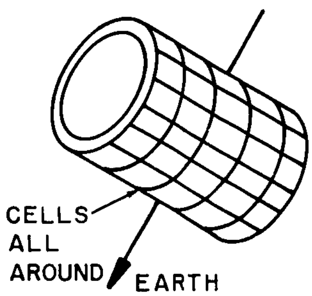
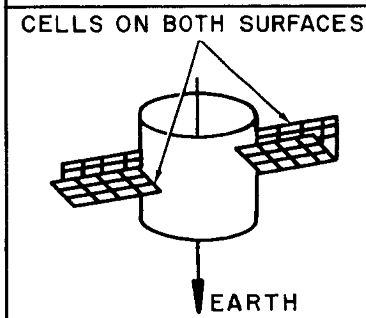
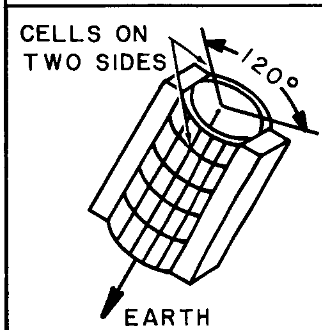
CONFIGURATION	TYPE OF ARRAY	AREA PENALTY FACTOR	SOLAR CELL AREA FACTOR
	ONE-AXIS ORIENTED ARRAY	1	1
	SPIN STABILIZED	π	π
	FIXED ARRAY (90° PANELS)	1.66	3.32
	BODY MOUNTED	3.28	3.28

Figure 4-10. Solar Cell Arrays for Gravity Gradient Spacecraft

sensor used to provide a synchronization signal for the meteorological sensors and the control torquers. The Polaris tracker signal is fed into an error amplitude discriminator to provide a dead zone which prevents control action until a specified error magnitude is exceeded. This level is set to about 1° so that corrective action is not taken in response to sinusoidal external disturbance torques. Velocity error and station keeping connections are provided by pulsing a single radial gas jet whose thrust axis goes through the center of mass. The pulses are synchronized by the Earth sensor signals.

The precession rates of the vehicle's spin axis will be approximately 5×10^{-5}°/sec for normal solar disturbance torques. By using the Polaris tracker, the control system will maintain the spin axis within 1° of the Polaris line (which is approximately 55 arc minutes from the North celestial pole).

5. Power Supply

Three basic configurations were examined for the minimum performance satellite:

(1) Spin stabilized

(2) 3-axis stabilized

(3) Gravity gradient

Power supply systems were sized for the three. In addition, certain alternate solar arrays were investigated for gravity gradient.

Initial investigation established that the power level would be over 100 W. This determined that an oriented array would be used on the 3-axis stabilized satellite, and eventually determined the surface area of the spin stabilized configuration.

The peculiar requirement of the gravity gradient satellite, that both of the separate sections be balanced in projected area, led to complexity in deployed panels. Therefore, solar arrays of body mounted cells, and fixed deployable panels were compared with oriented arrays, to arrive at the best compromise between weight and complexity. Figure 4-10 compares all of the arrays considered for this satellite. The comparison is on the basis of number of cells required and total area rather than on weight since the specific weight of oriented arrays varies with their area (see Figure 4-11). It must be noted that a body mounted solar array pays an additional weight penalty over an oriented array for thermal control. There is no good way to conduct the heat away from the exposed array. Consequently, some means (probably a circulating fluid system) must be used to maintain the cells at a reasonable temperature.

Tables 4-5 through 4-7 summarize the estimated power requirements of the various configurations. It has been assumed that no data will be acquired or transmitted during occult periods.

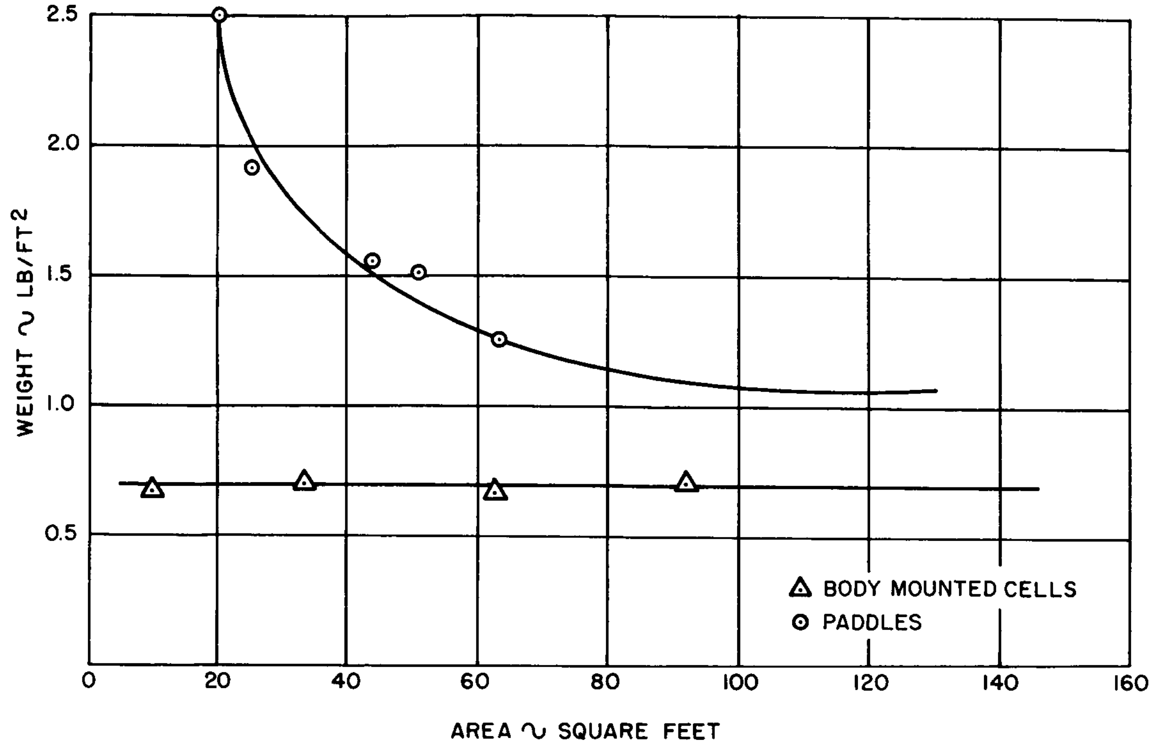


Figure 4-11. Solar Panel Weight

TABLE 4-5
POWER SUMMARY - SPIN STABILIZED MINIMUM CAPABILITY VEHICLE

Primary battery	8.5 lb
Secondary battery	6.2 lb
Solar cell area	33.6 lb (48 ft^2)
Regulator-Selector	12.0 lb
Total	54.3 lb

System Requirements	Acquisition (W-hr)	Track Average (W)	Track Peak (W)
Attitude Control	21.2 x 1 hr	31.7	56
Communications	22 x 6 hr	29.5	42
Data Handling	22.4 x 6 hr	24.4	24.4
Power Supply	7 x 6 hr	7	7
Sensor Equipment	0	7	15
	339.6	99.6	144.4

TABLE 4-6
POWER SUMMARY - 3-AXIS STABILIZED MINIMUM CAPABILITY VEHICLE

Primary Battery	9.7 lb
Secondary Battery	21.5 lb
Solar Array	44.0 lb (26.7 ft^2)
Regulator-Selector	12.0 lb
Solar Paddle Drive	3.0 lb
Total	90.2 lb

System Requirements	Acquisition (W-hr)	Track Average (W)	Track Peak (W)
Altitude Control	79 x 1 hr	105	241
Communications	22 x 6 hr	29.5	42
Data Handling	22.4 x 6 hr	24.4	24.4
Power Supply	6 x 6 hr	7	7
Sensor Equipment	0	7	15
Total	387.4	172.9	329.4

TABLE 4-7
POWER SUMMARY - GRAVITY GRADIENT MINIMUM CAPABILITY VEHICLE

Primary battery	9.5 lb
Secondary battery	21.4 lb
Solar cell	66 lb (26.7 ft^2) (Two oriented arrays)
Regulator-Selector	12 lb
Total	108.9 lb

System Requirements	Acquisition (W-hr)	Track Average (W)	Track Peak (W)
Attitude Control	71 x 1 hr	102.5	236
Communications	22 x 6 hr	29.5	42
Data Handling	22.4 x 6 hr	24.4	24.4
Power Supply	7 x 6	7	7
Sensor Equipment	0	7	15
Total	379.4	170.4	324.4

6. Thermal Control

The spin stabilized configuration is best suited for a passive thermal control system. The spinning action imposed on the satellite for reasons of stabilization constitutes a semiactive thermal control system with no penalty for complexity.

The proposed configuration is shown in Figure 4-1. Equipment is distributed radially in a toriod that makes up the basic structural section of the satellite. Attached to each end of the toroid are the skirts supporting the required number of solar cell modules. Figure 4-12 shows the temperature of the satellite while in orbit. During the ascent to synchronous orbit, the satellite may be exposed to various thermal conditions, depending on time of launch, trajectory, etc. To provide flexibility of installation and safety for sensitive parts that may be exposed, suitable insulation and coatings will be used to prevent overtemperature during maximum solar heat inputs. Local internal heating will be provided to prevent temperatures from dropping to levels that could cause permanent damage.

It will be noted in Table 4-1 that a 16 lb weight penalty is assessed against the gravity gradient satellite for body mounted solar cells. This is the estimated weight of a semi-active cooling system using a pump and circulating fluid. A completely passive system with a sufficient quantity of heat of fusion material to maintain the solar cells at approximately 100°F would weigh 182 lb. Volume 5 contains a detailed description of thermal problems associated with this type of spacecraft.

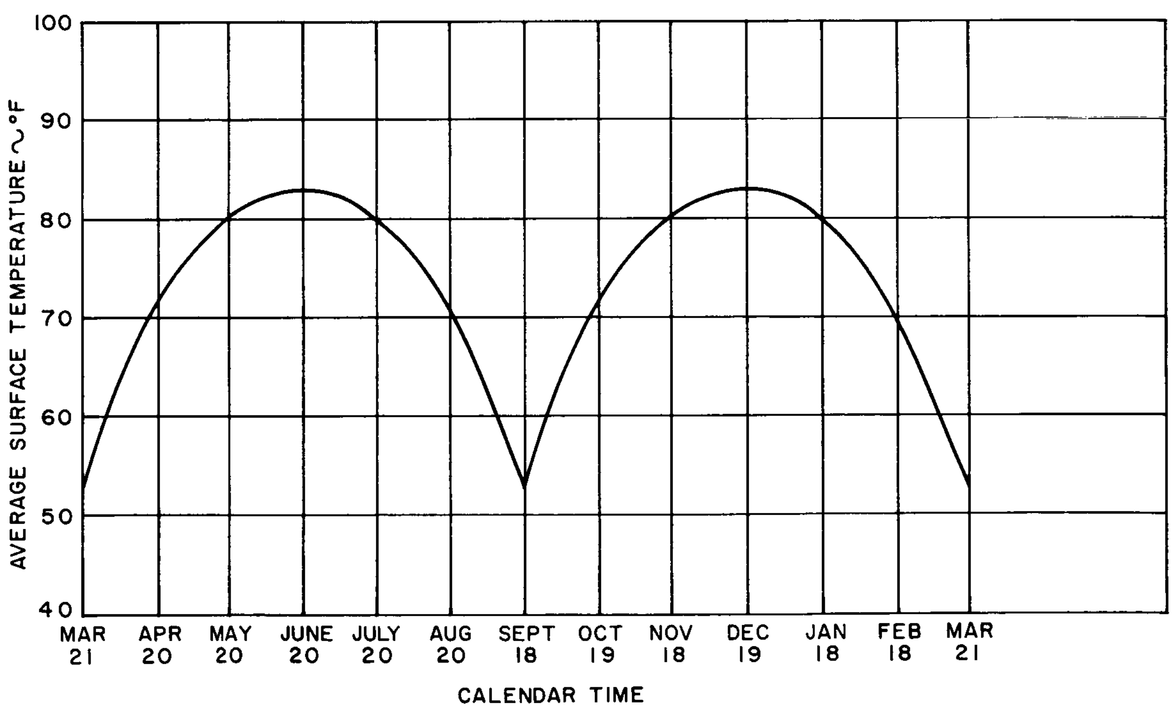


Figure 4-12. Temperature Variations in Orbit

7. Structure

The structural arrangement of the satellite is shown in Figure 5-1. As pictured, the configuration consists of an outer cylindrical skin, a lower conical skin, and a torodial equipment bay consisting of end bulkheads, concentric full depth circular beams and radial beams required for stiffening and support of the equipment. The toroid is divided into four equipment bays with external access doors. The solar cells are attached to separate skirts at each end of the center toroid and fastened to the basic structure. These skirts are sized to accommodate the required number of solar cells. They are not designed to take any structural loads other than their own weight.

This structural arrangement was selected to isolate the solar cell substrate from peak or concentrated loads applied by the apogee motor or transmitted through the adapter from the primary booster.

8. Propulsion

The intent of the study was to use the Thor-Delta as the launch vehicle. The proposed vehicle consists of three sections:

(1) first-stage liquid propelled Thor, DM-21 with 170,000 lb sea level thrust

(2) second stage liquid propellant motor (second stage Vanguard) AJ10-118 with 7700 lb thrust

(3) solid propelled spin stabilized Altair of 2800 lb thrust. This booster can put approximately 100 lb in a 22,240 nm elliptical orbit. It does not have the capability of attaining an equatorial orbit, because the upper stage does not have the ability to orient the payload for proper vectoring of the apogee kick.

Because the Thor-Delta is not adequate for the payload, other boosters, such as the Thor-Ad were investigated, but with no success. The next standard launch vehicle is the Atlas-Agena, which makes the mission decidedly uneconomical. One possibility of using the payload capacity of the Atlas-Agena might be to "piggy-back" another satellite or scientific experiment.

The apogee kick motor for this satellite was estimated from Figure 2-18. Using a propellant, I_{sp}=290 sec, and propellant fraction of 0.91, the apogee motor weight is approximately 265 lb.

E. PROBLEM AREAS

The primary problem associated with this satellite concerns the mechanism for image compensation. Vehicle rotation must be accurately determined and image despin mechanism precisely synchronized. Although the sensors selected for this configuration do not have an image compensation feature, their performance could be enhanced considerably by this feature.

Other problems can be defined as:

(1) Development of sensors with adequate unattended life

(2) Development of mechanism to counteract wobble of the spin axis due to damage or equipment shift

(3) Development of means to prevent direct impingement of Sun rays on sensitive detectors

(4) Means of acquiring the correct star (Polaris) with the star tracker

(5) Development of horizon scanners with the properties required for this mission

SECTION 5 - GRAVITY GRADIENT SPACECRAFT

A. CONFIGURATION

A group of satellites has been configured to use the effect of gravitational force for Earth orientation. They offer the possibility of good pointing accuracy and low oscillation rates, thereby providing good sensor platforms. Natural forces can be used to simplify the attitude control system at the expense of weight, and complexity of structure and power supply. A discussion of the principles of gravity gradient follow in subsection B.

The gravity gradient satellite is in effect a special case of the 3-axis stabilized satellite, hence the mission capability is the same as comparable configurations analyzed previously in this volume. The information capability of the gravity gradient stabilized satellite is approximately the same as that of the 3-axis stabilized satellite because the sensor equipment is the same. The gravity gradient satellite excells in platform stability primarily because it uses an excellent rate gyro and is independent of horizon sensors and their limitations. It has the same limits on yaw oscillation rate and yaw orientation as the 3-axis stabilized vehicle, since it uses the same means for sensing and control. Pointing accuracy is strictly a function of vehicle inertia. It compares well with the 3-axis spacecraft, and when sized appropriately, has sufficient control to counter random disturbances such as solar flares.

To permit a reasonable appraisal of the gravity gradient satellite, it is compared with the 3-axis stabilized satellite of like sensor capability. Such a comparison does not make allowance for the superior stability of the gravity gradient platform which is difficult to match with active 3-axis control having horizon scanners in the control loop. Table 5-1 details the weights of the two configurations being compared and points out the salient penalties of both.

Figure 5-1 is an overall block diagram that shows the relationship of the various subsystems.

Very briefly, pointing accuracy, the inherent inclination for the satellite to line up with the local vertical to the Earth, and resistance to disturbing torque, induced by solar flux, are functions of the moment of inertia of the vehicle and gravitational force. At synchronous altitude, the gravitational force is low, thereby requiring the vehicle to have a high moment of inertia. To attain these moments of inertia, the proposed spacecraft is divided into two equal symmetrical masses; after synchronous orbit injection, the masses separate to the required distance.

Equal masses were chosen because they lead to the minimum separation distance. Figure 5-2 shows the increased separation distances required when the masses are unequally distributed. Symmetrical shapes are desired because they minimize the torque created by solar radiation impinging on unequal areas. Any unbalanced torque can only be counteracted by an increase in vehicle inertia. Figure 5-3 indicates the relationship between vehicle length and unbalanced torque for a constant deviation angle (ϕ_{SS}). Figure 5-4 indicates the sinusoidal pitch rate of the vehicle due to the influence of solar flux.

TABLE 5-1
WEIGHT COMPARISON
GRAVITY GRADIENT VERSUS 3-AXIS STABILIZED
MEDIUM CAPABILITY SATELLITE

Item	3-Axis	Gravity Gradient
Sensor Equipment	83.0	83.0
Communications	22.0	22.0
Data Handling	29.5	29.5
Power Supply Total	127.7	a. 188.1 b. 146.7 c. 184.2
Batteries	39.7	
a. Body Mounted Solar Cells		a. 59.7
b. Oriented Solar Paddles		b. 39.7
c. Fixed Solar Paddles		c. 39.7
Charger	12.0	12.0
a. Body Mounted Solar Cells		a. 116.4
b. Oriented Solar Paddles	76.0	b. 95.0
c. Fixed Solar Paddles		c. 132.5
Attitude Control and Station Keeping	103.0	82.2
Wiring and Harness	22.0	29.0
Separation Penalty (28 ft)		6.0
Thermal Control	10.0	
a. Body Mounted Solar Cells (passive + fluid loop)		a. 26.0
b. Oriented or fixed solar paddles (passive)		b. or c. 10.0
Structure Total	108.0	127.0
#1 Body Break for Separation Penalty,		12.0
Separation Boom and Inflation Device		7.0
or #2 Separation Boom and Positive Drive Device		or 19.0
Despin Mechanism	18.8	17.6
Satellite Weight:	524.0	
a. Body Cells		a. 604.4
b. Oriented Paddles		b. 547.0
c. Fixed Paddles		c. 584.5
+ Adapter	42.0	42.0
+ Propellant	670.0	
a. Body Cells		a. 1094.0
b. Oriented Paddles		b. 657.0
c. Fixed Paddles		c. 700.0
+ Apogee Motor Case:	75.0	
a. Body Cells		a. 122.0
b. Oriented Paddles		b. 74.0
c. Fixed Paddles		c. 78.0

TABLE 5-1 (continued)

Item	3-Axis	Gravity Gradient
Total Payload Weight:		
(At Booster Take-off)	1311.0	
a. Body Cells		a. 1862.4
b. Oriented Paddles		b. 1320.0
c. Fixed Paddles		c. 1404.5

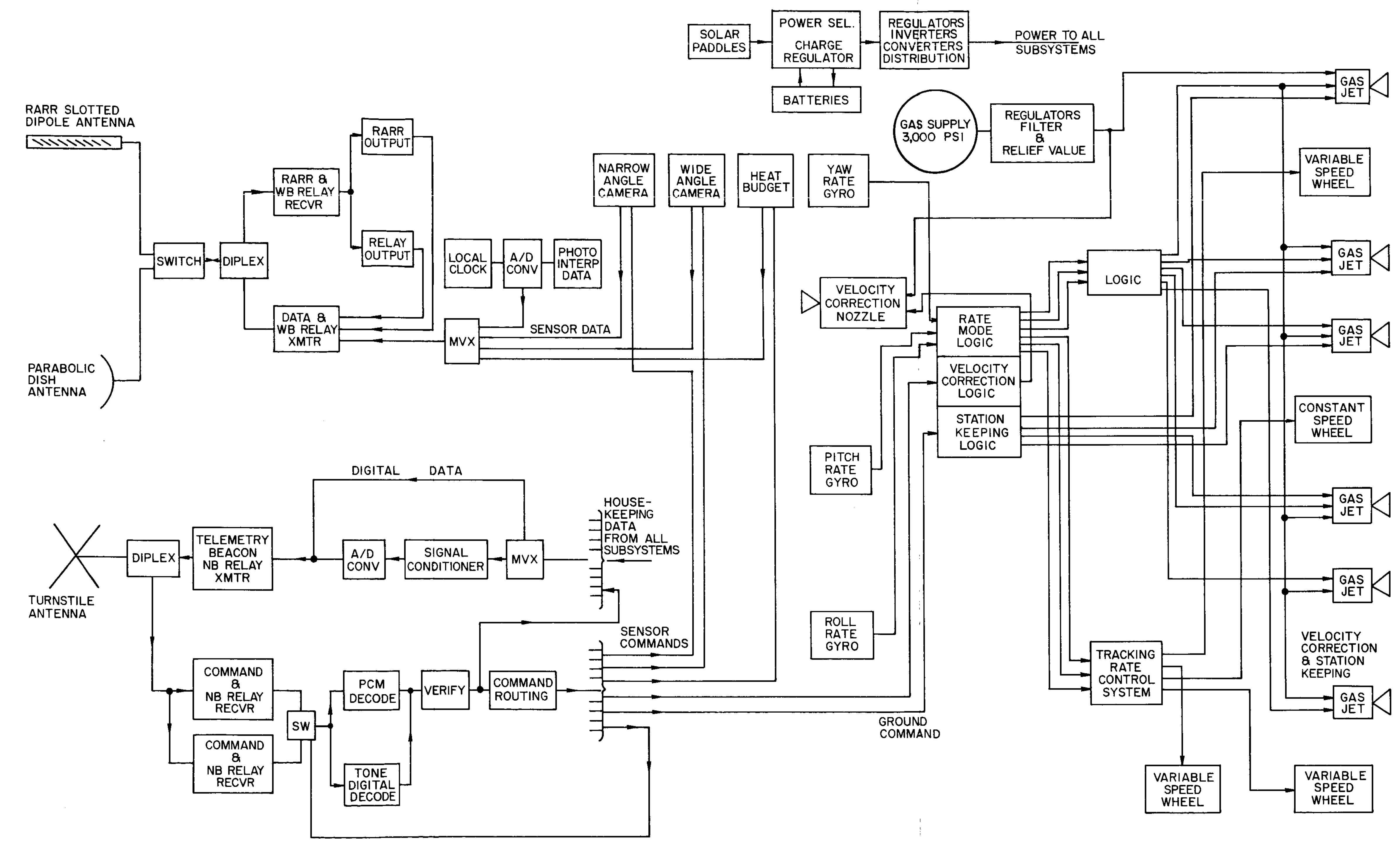


Figure 5-1.
Gravity Gradient Spacecraft Subsystems

The models for this investigation were basically drum-like forms covered by solar cells. Because these bodies are stabilized and oriented in space, some of the solar cells are very inefficient. A better arrangement is one using only two sides of the satellite (those that alternately face the Sun) covered with solar cells. The remainder of the satellite is then designed to form two space-oriented panels that can be used to mount equipment and use the shadow box thermal control technique formerly described. The solar cells must be thermally controlled by an active system to maintain reasonable temperature limits.

Body mounted solar cells do not deliver the required amount of power because the nose cone envelopes of the proposed launch vehicles (except the maximum capability satellite) do not permit sufficient surface area to be developed. An additional factor is that the effective area of the panels changes due to the rotation of the satellite in relation to the Sun. For substantial periods of each day they yield little or no power due to the highly incident angle of the Sun rays. To provide for the deficiency of solar power occurring twice daily, either more batteries are required, or a higher depth of discharge must be permitted. One solution to this problem is to add to the satellite external solar arrays of the proper size in place of the body mounted cells. Oriented arrays were compared with fixed arrays and found to present a substantial advantage. However, any solution involving external arrays leads to a further complexity of having to duplicate arrays on both satellite halves. A single array on one section increases the unbalanced disturbance torque approximately two orders of magnitude, which in turn leads to extreme separation distances. Figure 5-5 illustrates two conceptions of the medium capability gravity gradient satellite, both with body mounted solar cells and an oriented array.

There will be sizeable velocity corrections required due to orbit injection errors. The satellite must be correctly oriented before reaction jets can be actuated to effect the necessary corrections. If it is assumed that the gas and control systems are kept in one section of the satellite, the jet thrust must be kept small; or the effect of the jet acting as a disturbance torque will increase deviation from vertical, or hangoff, to an extent that would render sensors and jets useless. If the thrust was of a level that overpowered gravity gradient restoring torque, the satellite would merely pinwheel. As an example, the maximum force tolerable on a single jet on the medium capability (symmetrical) satellite is 9.8×10^{-7} lb, which would cause 30° hangoff of the satellite. If the satellite-Earth orientation were to take only one hour, the jet force would have to be applied for 10^7 sec to correct the nominal injection error of 40 ft/sec. It becomes obvious that the small single jet is not practical. Higher force jets must be supplied in both halves of the satellite, with duplicate gas supply systems, because it is not considered feasible to pump gas through flexible lines from one section to the other. The duplication of the gas system is obviously an additional weight penalty.

The problem is reduced, but not to a tolerable extent, by location of jets on the asymmetrical configuration in Figure 5-6. This illustration shows an unsuccessful attempt to eliminate some of the problems encountered in trying to minimize satellite size. It will be seen that the satellite is not symmetrical nor are the two ends of equal mass. This was a deliberate effort to bring the cg closer to one section and reduce the moment arm sufficiently to use one jet. This particular configuration has been sized to obtain a pointing accuracy of approximately 1/2°, under influence of normal solar pressure. It will deviate

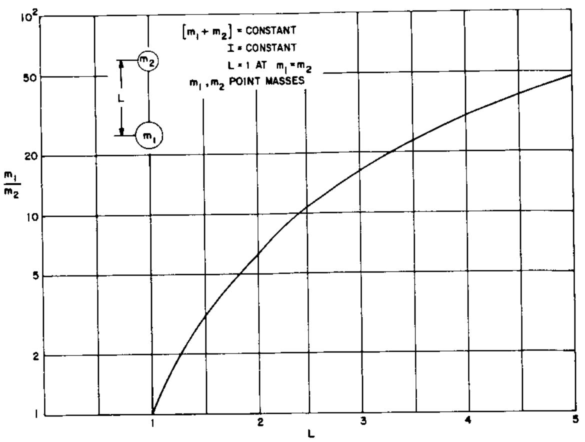


Figure 5-2. Separation Distance vs Mass Ratio

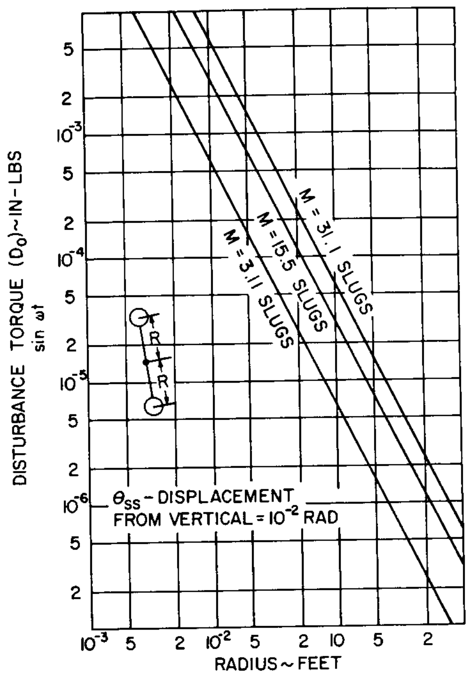


Figure 5-3. Disturbance Torque vs Angular Displacement from Vertical

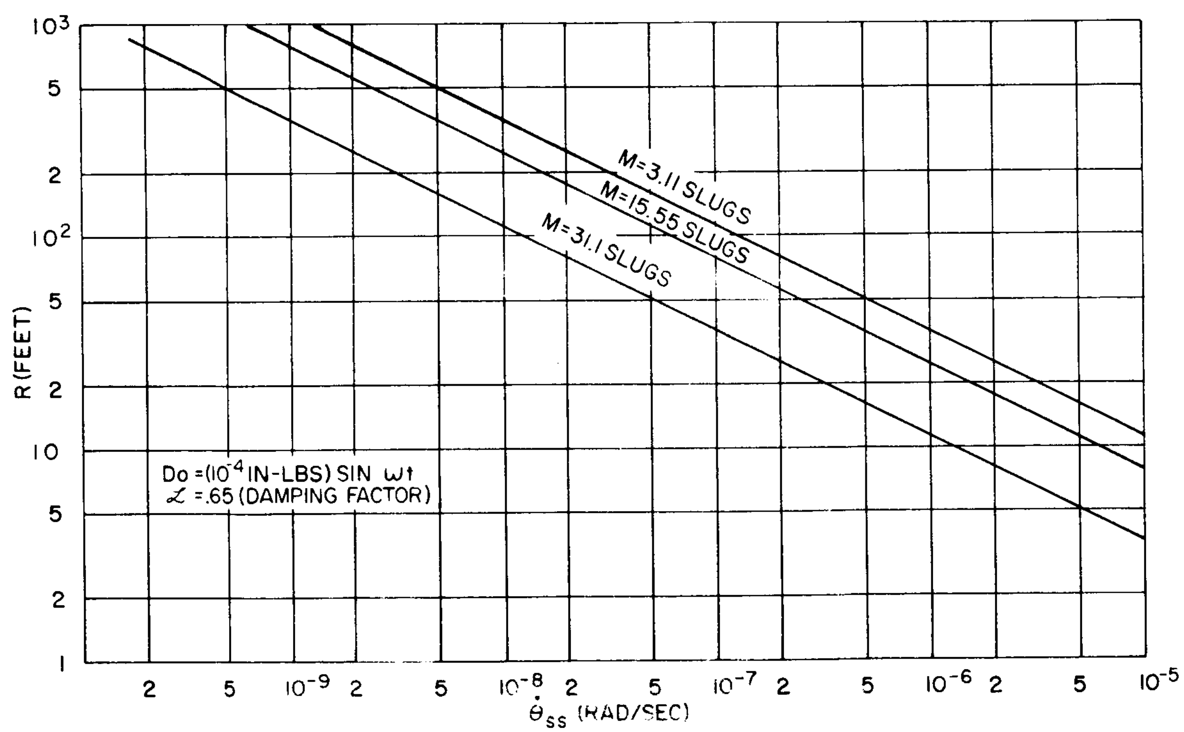


Figure 5-4. Pitch Rate Due to Influence of Solar Flux

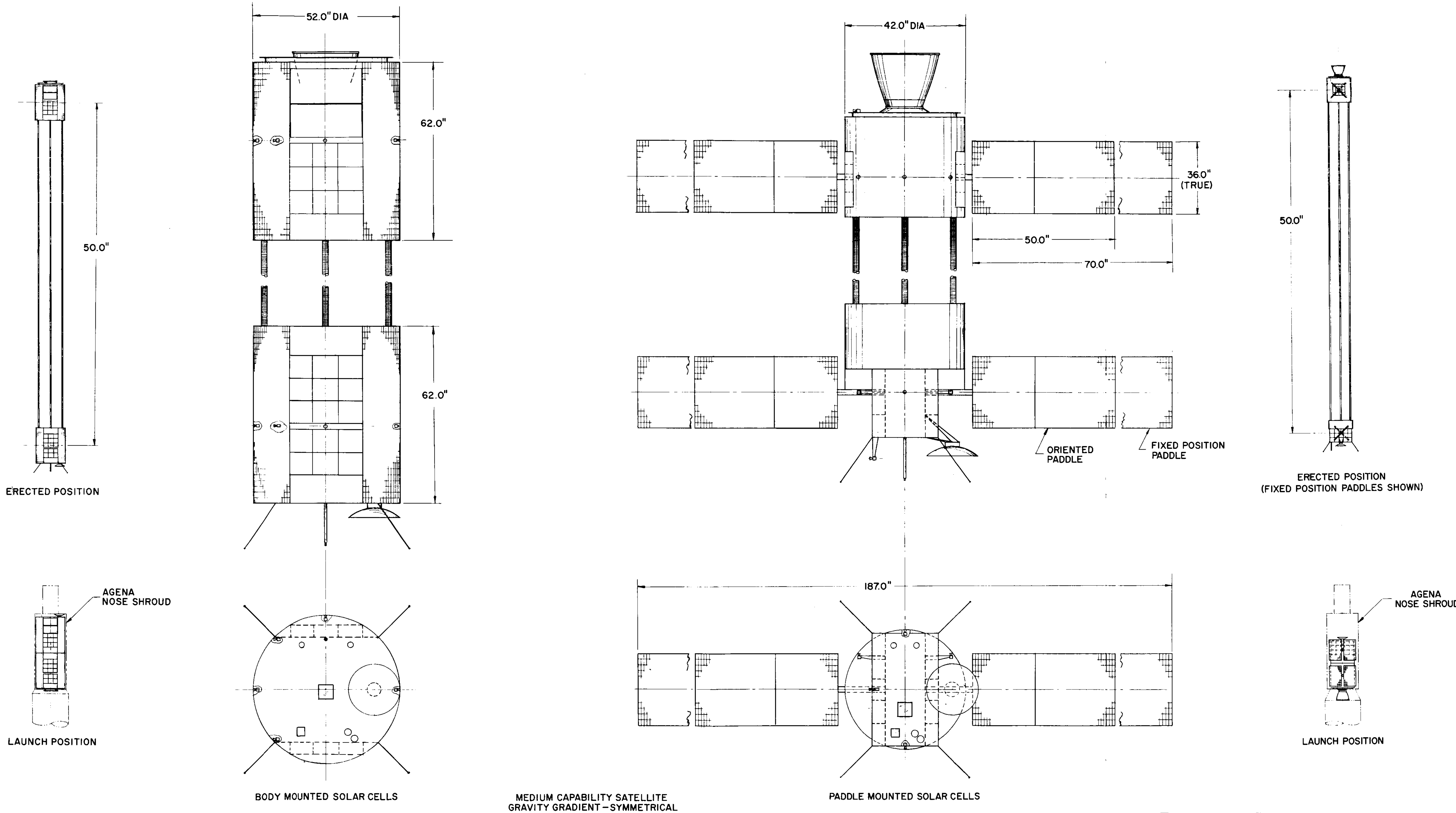


Figure 5-5. Gravity Gradient Spacecraft

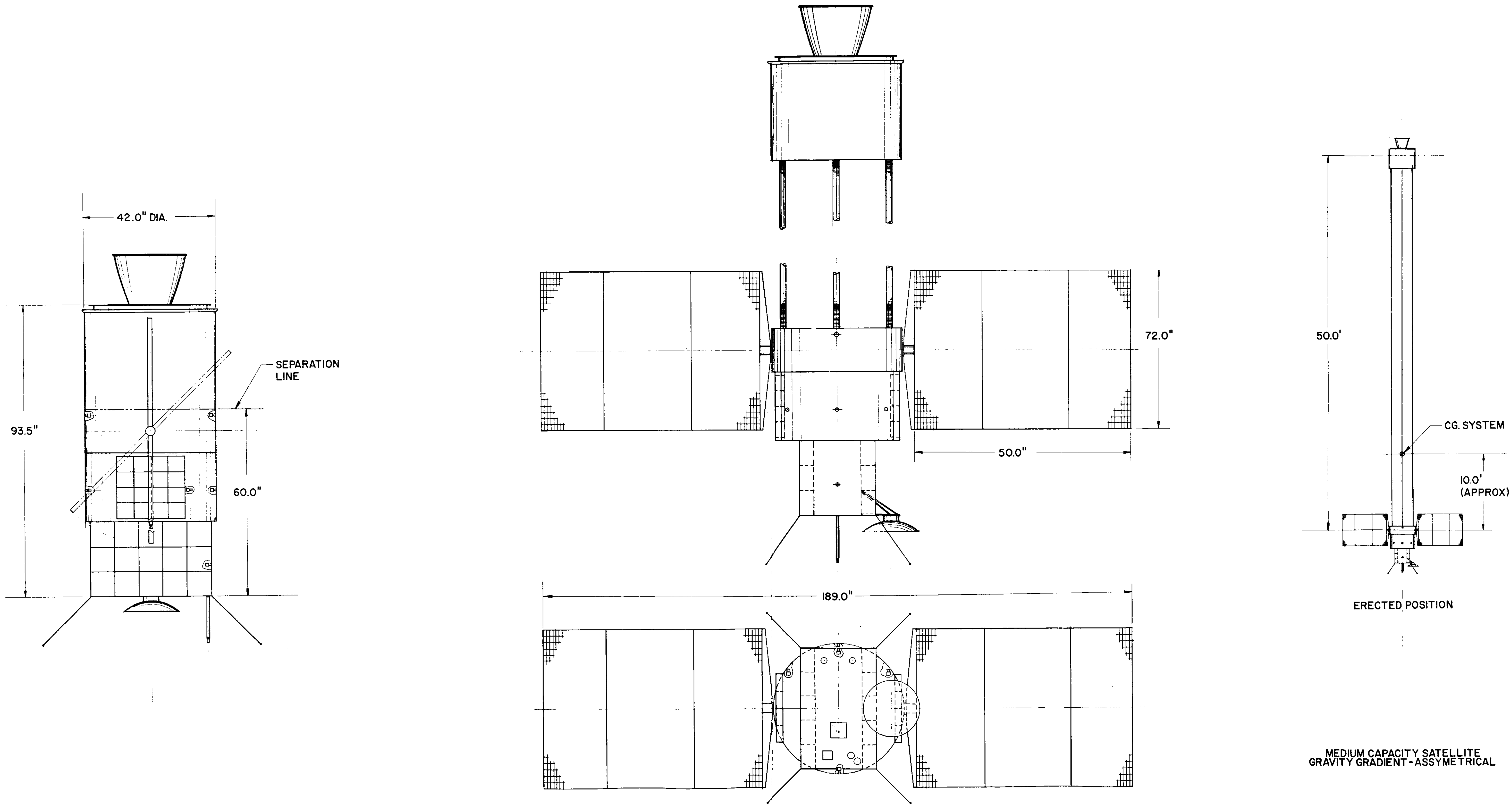


Figure 5-6. Asymmetrical Gravity Gradient Vehicle

from the local vertical under normal Sun pressure approximately the same amount that a symmetrical body of equal mass would under the influence of a solar flare (roughly two magnitudes of force greater). Active control is required to counter solar flare which would produce approximately 30° deviation. All equipment has been located in one area. The balance of the satellite consists of the empty apogee motor case and the adapter ring. The cg of the configuration is still too high to permit proper location of the reaction jets.

Interconnecting circuits must be run between the satellite parts whenever the two sections contain operating equipment. Careful location of the equipment can minimize the number of circuits, but since control functions are needed in both sections, circuits cannot be completely eliminated.

Figure 5-7 illustrates the size of symmetrical satellites of nominal weights to obtain similar pointing accuracies and rates.

B. GRAVITY GRADIENT THEORY

The torques applied to a satellite due to the gravitational field of the Earth (considered to be spherical) are:

$$T_x = \frac{-3\ GM}{R^3} (I_y - I_z) \text{ Cos}^2 \theta \text{ Sin } \psi \text{ Cos } \phi$$

$$T_y = \frac{-3\ GM}{R^3} (I_x - I_z) \text{ Sin } \theta \text{ Cos } \theta \text{ Cos } \phi$$

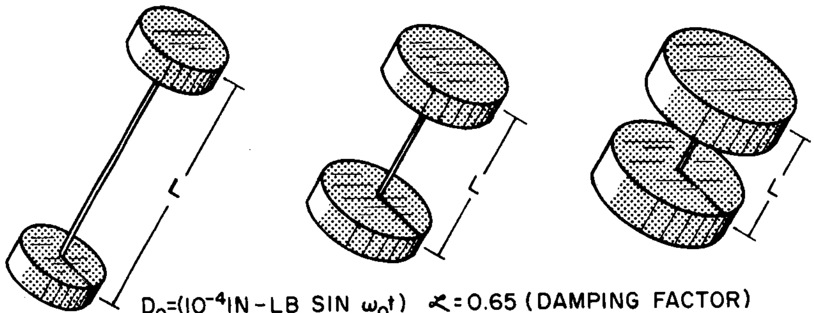
$$T_z = \frac{-3\ GM}{R^3} (I_y - I_x) \text{ Sin } \theta \text{ Cos } \theta \text{ Cos } \phi$$

where:

where:	G	=	Universal gravitational constant
	M	=	Earth mass
	R	=	Position vector of satellite mass center, with respect to Earth
	I_x	=	Satellite moment of inertia about body x-axis
	I_y	=	Satellite moment of inertia about body y-axis
	I_z	=	Satellite moment of inertia about body z-axis

θ, ϕ, ψ are the Euler angles relating the satellite body axis orientation to the orbital reference frame as defined by Figure 5-8. All products of inertia have been neglected, i.e., it is assumed that the satellite body and control axis are aligned. For small angular motions, such as are encountered in the Earth tracking mode, the above expressions become:

$$T_x = -3\ W_o^2 (I_y - I_z)\ \phi$$



$D_0 = (10^{-4}$ IN-LB SIN $\omega_0 t)$ $\propto = 0.65$ (DAMPING FACTOR)

	100#	500#	1000#
$\theta = 1°$ $\dot{\theta} = 0.0068°$/SEC LENGTH (L)	$I_{xx} = 3300$ SLUG FT^2 64'	$I_{xx} = 3900$ SLUG FT^2 28'	$I_{xx} = 4600$ SLUG FT^2 21'
$\theta = 0.5°$ $\dot{\theta} = 0.00315°$/SEC LENGTH (L)	$I_{xx} = 6500$ SLUG FT^2 90'	$I_{xx} = 8000$ SLUG FT^2 42'	$I_{xx} = 8600$ SLUG FT^2 30'
$\theta = 0.1°$ $\dot{\theta} = 0.00074°$/SEC LENGTH (L)	$I_{xx} = 31,500$ SLUG FT^2 200'	$I_{xx} = 29,800$ SLUG FT^2 84'	$I_{xx} = 37,100$ SLUG FT^2 66'

Figure 5-7. Symmetrical Gravity Gradient Vehicle Comparison

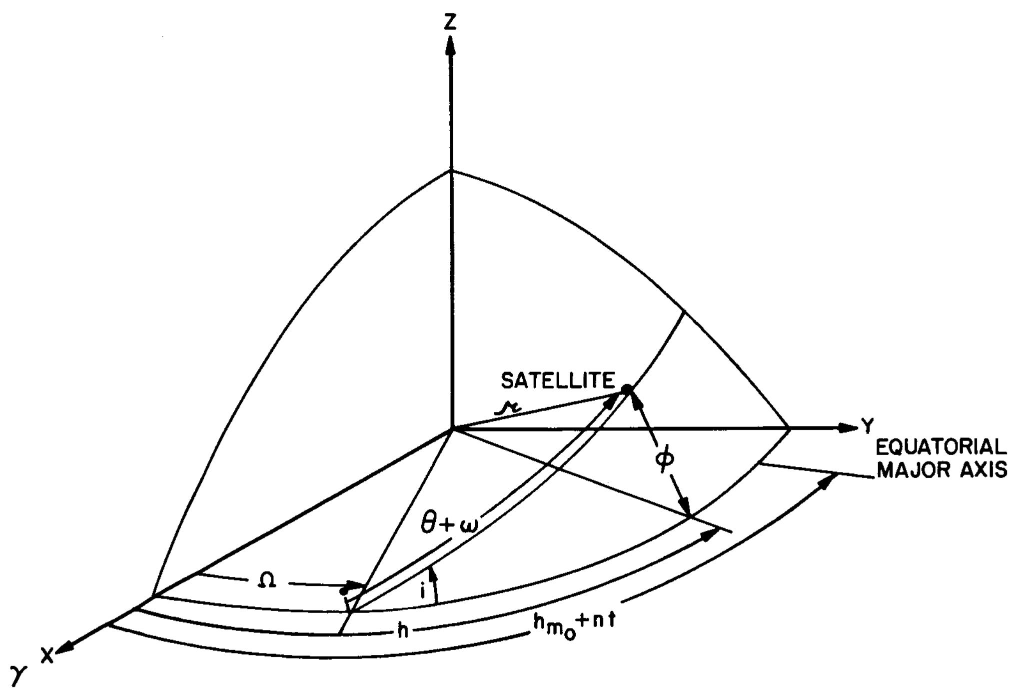


Figure 5-8. Orbital Reference Frame

$$T_y = -3\, W_o^2 \,(I_x - I_z)\, \theta$$

$$T_z = 0$$

for a circular orbit. These expressions show that, for applications where the assumptions made are valid, the effect of the Ensho gravitational field gradient is to provide a spring type restoring torque. Due to the nature of this torque, no damping is present, however, so that either passive or active damping must be provided.

C. ATTITUDE CONTROL

A block diagram for the gravity gradient control system is presented in Figure 5-9. Although the control for this type of satellite may be instrumented very easily for the Earth tracking mode, sufficient additional complexity is required for acquisiton so that it has only two basic advantages from a control viewpoint. These are the elimination of all attitude sensors for Earth tracking, and very low body rates while Earth tracking.

The Earth tracking mode will be discussed first, followed by the acquisition system discussion.

1. Earth Tracking Mode

The Earth tracking control system consists of three variable speed reaction wheels, one constant speed wheel, and three rate gyros. A gas system is not required for momentum dumping, but is provided to perform the station keeping and initial velocity correction functions.

Restoring control torques in pitch and roll are provided by differential gravity if the satellite's long axis is misaligned with the vertical. Damping is provided by reaction wheel torquers which are excited by vehicle pitch and roll body rates. It is clear that no reaction wheel dumping is required if the vehicle momentum is essentially zero with respect to the orbital reference frame at the time the reaction wheel rate damping system is energized. The pitch rate gyro output signal must be biased at a rate equivalent to orbital angular velocity so that the pitch reaction wheel speed is unresponsive to this constant rate.

Ground commands are provided for station keeping and velocity correction functions. The logic blocks perform similar functions to those of the 3-axis stabilized system as does the priority logic block. The rate mode logic selects one of two rate systems, depending on whether the system is in the acquisition or the Earth tracking phase. The Earth tracking rate system provides proportional control using reaction wheels, while an on-off gas system is used for acquisition rate control.

Selection of the Earth tracking rate control system energizes the constant speed wheel for gyro compass control of yaw attitude. During the time this wheel is accelerating up to speed, a pitch reaction torque will exist. Because the pitch variable speed wheel cannot store momentum equivalent to that of the constant speed wheel, and gravity gradient torque is too small to restrain vehicle

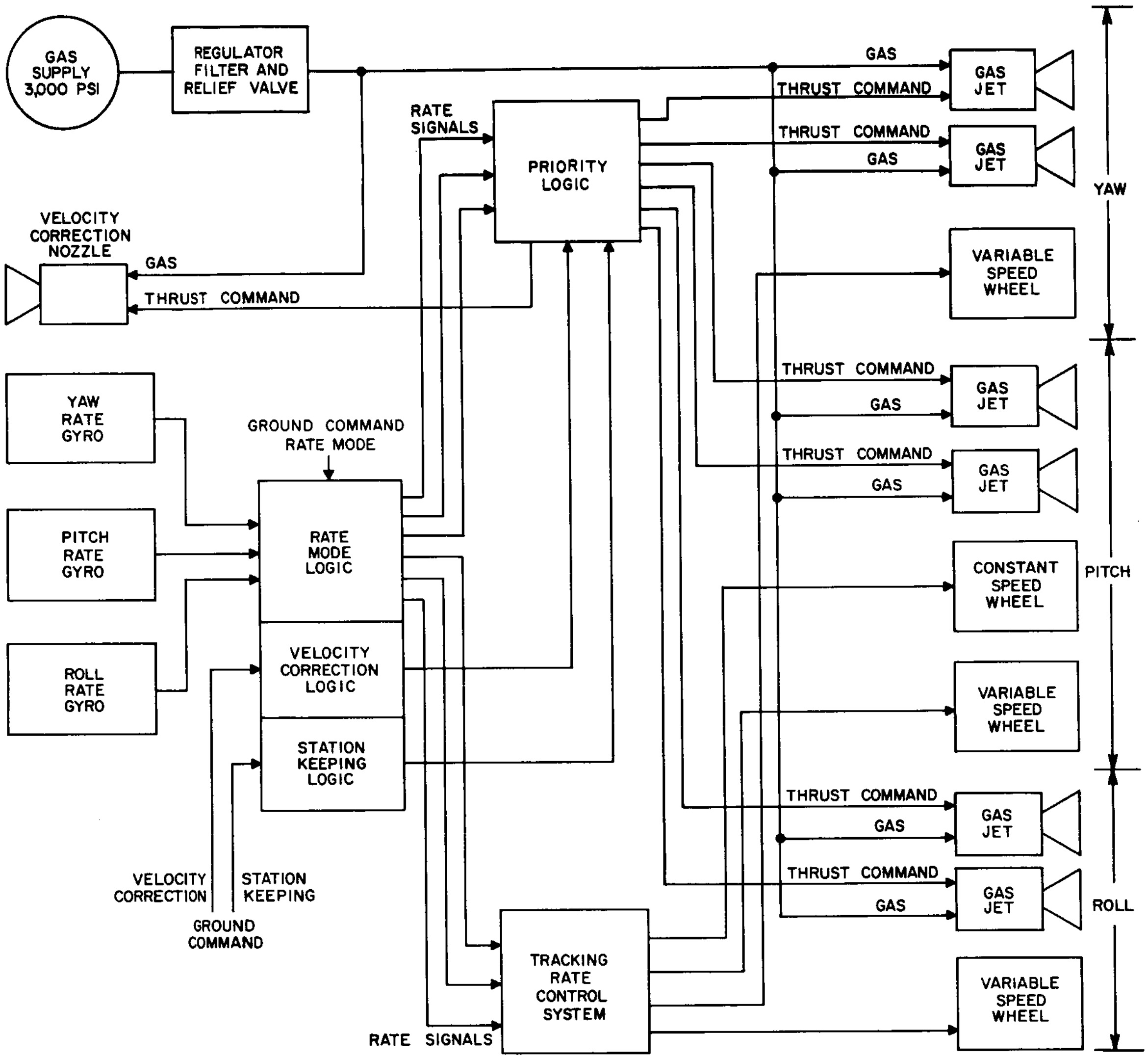


Figure 5-9. Gravity Gradient Control System

motion, the pitch gas system must be used during this period of wheel speedup. When wheel momentum has reached its steady state value, rate damping is once again accomplished for the pitch axis by means of the pitch reaction wheel.

2. Earth Acquisition Mode

As discussed in Section 2.E.7 of Volume 4, the gravity gradient spacecraft would consist of a simple control system for Earth tracking which would provide effective rate control to very low rates. However, the Earth acquisition requirement adds additional complexity to the overall gravity gradient system. The addition of a gas system is required to perform the acquisition function, in particular for active despin and during initial alignment. Thereafter, only the reaction wheels and rate gyros are used for damping, while restoring torques are provided by gravity gradient and gyrocompassing.

The major portion of the acquisition phase of the gravity gradient spacecraft is accomplished prior to separation of the two masses which constitute the dumbell configuration. By completing the major portion of acquisition prior to erection, acquisition time would be considerabley less than if the vehicle had been erected. This is due to the considerably larger spacecraft moments of inertia when fully erected.

The first phase of acquisition would be passive despin of the spacecraft yaw axis to about 10°/sec. Rates about pitch and roll are considered to be 1.0°/sec. Following passive despin, active despin is accomplished by means of rate control systems using rate gyros and a cold gas system. The spacecraft body rates are reduced in all axes to about 0.01°/sec.

At this point in the acquisition sequence, it must be determined whether the section of the spacecraft containing the meteorological sensors is below the local horizontal. To ensure that the sensory section is oriented in the direction of the Earth during Earth tracking, that section of the spacecraft must be below the local horizontal prior to erection of the two mass spacecraft configurations. This aspect has not been investigated in great detail; however, the following possibilities exist for establishing its gross orientation.

(1) Wide angle photo taken from sensory package, and transmitted to ground. If Earth disc is viewed, orientation is grossly correct.

(2) Application of RF beacon techniques to determine coarse attitude information.

(3) Use of a hemispherical Earth sensor at high noon to obtain Earth direction information.

Ideally, the gravity gradient spacecraft's long axis should be aligned with the local vertical prior to separation of the two masses. By doing this, acquisition time is significantly reduced. This is because the spacecraft damped natural frequency when erected is $\sqrt{3}\ \omega_0$ which implies that considerable time would be required to align to the local vertical from an arbitrary initial attitude. Using one of the above attitude sensing techniques in conjunction with the on-board rate control system would prove beneficial in the reduction of acquisition time. The rate control system would be commanded from the ground to provide approximate alignment of the local vertical and the spacecraft long axis.

After vertical alignment has been completed, the constant speed pitch wheel would be brought up to speed to provide yaw alignment. Thereafter, the spacecraft is considered to be in the Earth tracking mode. At this time the velocity correction and station keeping functions may be initiated.

Table 5-1 represents estimated time period for each phase of the gravity gradient spacecraft acquisition.

TABLE 5-1
GRAVITY GRADIENT SPACECRAFT ACQUISITION TIME PROFILE

PHASE	DURATION	REMARKS
Passive Despin	0.17 min	100 RPM to 10°/sec
Active Despin	7 min	10°/sec to threshold; required if sensor package upside down
180°Rotation	3 min	
Vertical Alignment Prior to Erection	1.5 min	Rate controlled
Erection Time	5 min	
Final Vertical Alignment	15 min	
Yaw Alignment	60 min	Gyrocompass wheel speedup
TOTAL	91.67 min	

D. SENSORS AND COMMUNICATION

The sensor capabilities can be compared to those in the 3-axis stabilized satellites of comparable size. The increased stability of the gravity gradient satellite is used as a margin of safety to ensure that the sensor will work to its fullest capacity. The systems are sized and designed to perform an acceptable mission with platform stability of 3×10^{-3}°/sec. The 3-axis stabilization system can, with optimum filtering of horizon scanner noise, approach 10^{-4} °/sec rates. The gravity gradient satellites, sized for pointing accuracies of approximately $\frac{1}{2}$°, achieve normal rates of 10^{-5}°/sec.

Communications, telemetry, and data handling systems are directly comparable to the systems in medium and high capability 3-axis stabilized satellites.

E. STRUCTURE

The problems associated with the use of gravity gradient at synchronous altitudes have dictated the long dumbbell-like configuration used as models. Equipment installations and the basic structure are susceptible to normal treatment.

Separating and fixing the sections in space is a unique problem. Figure 5-5 shows the proposed configuration at launch, and after orbital erection. Three columns connect the two sections. Because these columns are quite long, unconventional means are required to store them during launch and later erect them in space without incurring undue weight penalties. Four methods that offer promise are described.

1. Rigidized Fabric

Westinghouse has developed a coated fabric material of continuous filament fiberglass (Figure 5-10). The material is woven on a stocking loom or braider and is in the form of a tube of suitable length after weaving. The tube is coated inside and out with a polymer composition which is flexible in Earth's atmosphere, but which rigidizes when exposed to space environment. The composition and matrix of the woven tube is such that it can be rolled or convoluted to fit in a compact area. The polymer composition can be varied to attain rigidization in from 15 minutes to one hour. One end of such a tube could be attached to each of the satellite sections. Using low pressure, the sections could be deployed in space. Once rigidized, the tube would maintain the sections at a fixed distance and position.

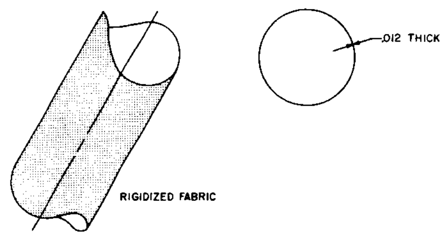


Figure 5-10. Rigidized Fabric Tube

2. STEM

DeHaviland Aircraft of Canada has developed a self-storing tubular extensible member (STEM) that may be used to separate the two satellite halves (Figure 5-11). This device consists of a metal tape that has been formed to a tubular shape when unrestrained, a suitable storage reel, and drive facilities. Initially, the tape is flattened and stored on the reel, similar to a carpenter's tape measure. When unreeled, it curls to form a tube. Little power is required to extend the tape and extension rate can be controlled by the drive.

3. Foamed Plastics

Goodyear Tire and Rubber Co. during their research on foaming plastics, formulated a compound that can be applied in thin layers, in plastic form, to a backing material (Figure 5-12). Under the influence of space environment, the compound foams and hardens. Approximately 1 mil of this plastic foams to 0.25 in. thickness. It is possible to apply this material to the inside surface of a tube of mylar, approximately 0.001 thick and 4 inches in diameter, and then coil the tube. The ends of the tube are attached to the satellite halves, and the halves deployed in space by the application of low gas pressure to the tube. Upon erection and exposure to space, the plastic foams and hardens, forming a thick walled tube. This tube would maintain the satellite halves in the required position.

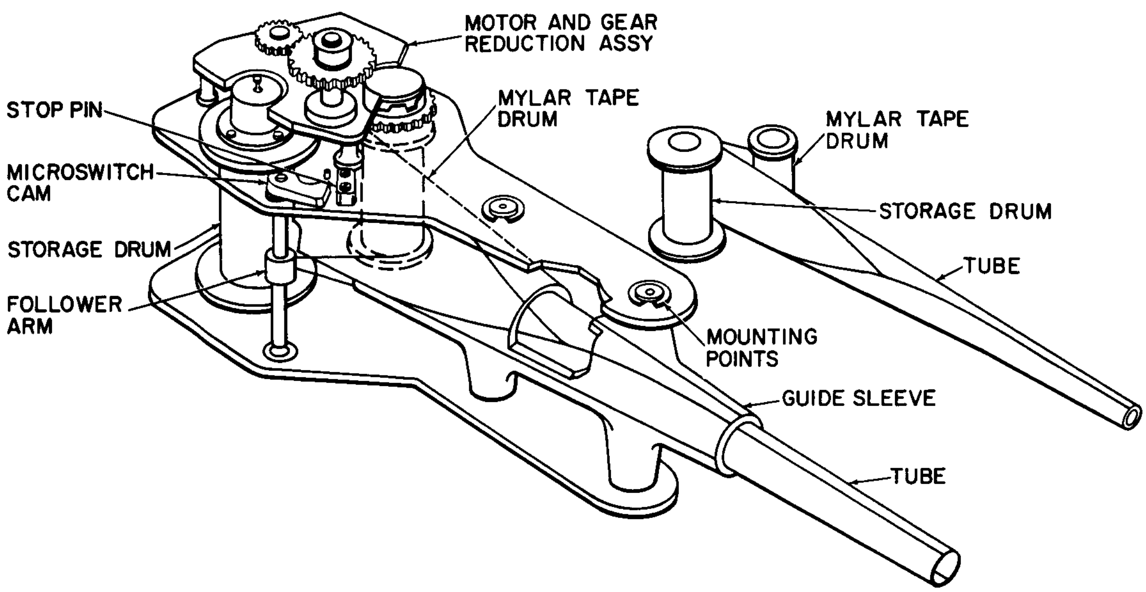


Figure 5-11. Metal Tape Extensible Tube

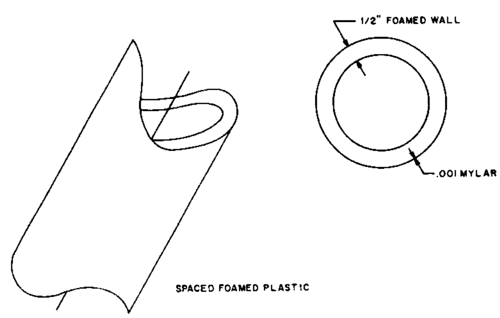


Figure 5-12. Foamed Plastic Tubes

4. Expanded Wire

Expanded wire structures have been extensively investigated by Goodyear (Figure 5-13). This is a system whereby a shape, usually cylindrical or spherical, is formed by a wire grid. The aluminum wires are about 0.005 inch in diameter and the grid is approximately 1/8in. x 1/4in. A plastic balloon of the same shape is bonded to the inside of the structure. The structure may then be collapsed into a compact size. Upon the application of low pressure, the inner balloon expands and returns the structure to its original shape. The pressure is adjusted so that the wire structure is stressed slightly above its elastic limit and takes a permanent set. If pressure is lost, the structure retains its shape. For space applications, the balloon may be made of a photolyzable material that decomposes in space environment. A column of correct length could be fabricated in the manner described, collapsed, and erected in space by air pressure to provide separation and fixity to the satellite halves. The resulting grid structure is thermally transparent.

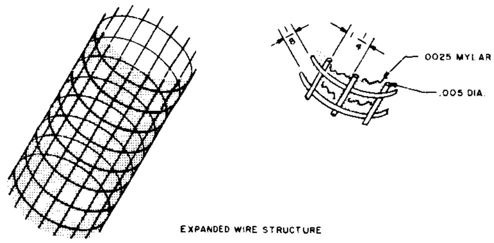


Figure 5-13. Expanded Wire Tubes

All of the tubes described have been developed for other than structural applications. In general, they have been required to support only their own weight under nominal g's or the negligible loads imposed by solar flux. However, their strength appears adequate to support the reaction loads expected from attitude control and station keeping. Table 5-2 compares the weights and strengths of the systems.

Problems exist concerning the erection of the satellite once orbit has been achieved and rough Earth orientation accomplished. The halves of the satellite must be freed (probably by using a shaped charge to destroy connecting structure) and then separated. For the sake of discussion, assume that an expandable wire grid structure is used for the interconnecting members. The structure is erected by the use of gas pressure which must be precisely controlled. The two halves have some velocity when they reach the correct separation distance. This velocity must be limited and the satellite sections damped to prevent damage to the interconnecting structure. During the period that the halves are separating, there can be no relative rotary motion between the halves. In fact, they must remain closely aligned in all directions until the interconnecting structure has "hardened," because this structure has no control over the satellite halves while they are in motion. Relative motion of the sections can lead to a misshapened satellite in which the sensor platforms are misaligned to a degree that severely impairs the mission. Further, substantial relative motion may result in inability to erect the satellite at all due to twisting of the structure. These considerations require that the satellite have zero body rates at separation, and that the separation be accomplished in a rigidly prescribed envelope. It must be noted that as the total mass of the satellite increases, the separation distances decrease. Positive mechanical means may be feasible for the 1000 lb class, eliminating many of the separation problems. Initial positioning of the two sections will be more positive if the erected tubes form a definite structure or truss. This type of structure would be more complex and would pose an added weight penalty.

Once the satellite is erected, the thermal gradients can cause appreciable warping in the interconnecting structure. This effect is most severe in the plastics. However, even in the expanded wire structure, which is thermally transparent and least subject to a temperature gradient, shading of parts of the structure because of satellite orientation can also lead to differential expansions. This action is similar to that produced by thermal gradients and produces the same effect.

Weight penalties must be assessed against the gravity gradient configuration for decoupling the two halves after orbit has been achieved, separating mechanism, power supply, and structure. Table 5-1 compares a 3-axis stabilized satellite to a gravity gradient oriented satellite with the same sensor capability.

TABLE 5-2
WEIGHT COMPARISON
GRAVITY GRADIENT INTERCONNECTING STRUCTURE

Type	Weight (lb/ft)	Separation Fixed Weight	Load Carrying Capacity lb-in.
Expanded Wire Grid (4.0 in. dia)	0.139	5.0	252
Rigidized fabric tube (4.0 in. dia)	0.139	5.0	47
STEM 0.9 in. dia.x0.005 t.	0.075	15.0	240
Foamed plastic tube (4.0 in. dia x 0.50 t)	0.087	5.0	--

F. POWER SUPPLY

Several problems exist in selecting the solar array arrangement for a gravity gradient satellite, posed in general by the requirement that the sections be of similar area. Applying solar cells directly to the body was the first method chosen for its simplicity, and was used on the models that established basic data.

Practical considerations of thermal control, however, dictate that only sections of the satellite be covered by cells. This is in itself no hardship because on the cylindrical oriented body, approximately 120° (total) of the cells are inefficiently used. These cells are replaced by space oriented panels that serve to maintain proper equipment temperatures. The rotation of the satellite about the pitch axis (one revolution per orbit) results in body orientation with the Sun that produces very little or no power for a substantial time twice each day. During this time, not to be confused with the occult period, all power must be supplied from storage batteries. This normal operating cycle dictates a larger battery supply because there is no intention of limiting satellite functions twice a day. In addition, to maintain the average power per day required, the body area must be sized to supply peaks 1.57 times the average. Possible solutions to the problem, involving fixed and oriented deployable panels, were investigated. Any type of deployable panel must be duplicated on both sections of the satellite, incurring a penalty in both weight and complexity.

Figure 4-11 compares the area of solar cells for oriented, fixed paddles, and body mounted cells. Table 5-3 summarizes the subsystem power requirements of the gravity gradient medium capability satellite. An oriented solar array was selected for the gravity gradient satellite to permit a close comparison with the 3-axis stabilized configuration (Table 2-2). It is again assumed that no data will be acquired during the occult period.

TABLE 5-3
POWER SUMMARY GRAVITY GRADIENT MEDIUM CAPABILITY

Primary Battery	18.2 lb
Secondary Battery	21.5 lb
Solar Cell	89 lb (Two 25 ft^2 arrays)
Regulator Selector	12 lb
Solar Paddle Drive (2)	6 lb (2 drives)
TOTAL	146.7 lb

System Requirements	Acquisition (W-hr)	Track Average (W)	Track Peak (W)
1. Attitude Control	71 x 1 hr	102.5	236
2. Communications	60 x 6 hr	50	75
3. Data Handling	31.1 x 6 hr	33.1	33.1
4. Power Supply	7 x 6 hr	7.0	7.0
5. Sensor Equipment	10 x 6 hr	77.1	131.0
	719.6	269.7	482.1

G. THERMAL CONTROL

Of the three types of stabilization systems considered in this study, the gravity gradient stabilized craft seems, at a first glance, to have the most difficult thermal control problems. First, because the vehicle does not spin, the self-regulating temperature properties of the spin stabilized configuration are absent (see Section 6 in this volume and Section 4 of Volume 5). Second, the possibility of applying the shadow box temperature control technique is ruled out because the area taken up by the shadow box is needed for solar cells.

However, if the available solar cell area on a cylindrical body is examined it is found that out of an available sector of 180° only a sector of 120° can be considered effective solar cell area. The rest of the semicircle contributes a negligible amount of power because practically all of the incoming solar energy is reflected. This means that by integrating two shadow boxes with the cylindrical structure as shown in Figure 5-14, the available effective solar cell area will not be reduced by an appreciable amount.

With this arrangement the gravity gradient configuration becomes identical to the 3-axis stabilized configuration discussed in Section 4 of Volume 5 and in Section 2 of this volume. The only difference is that in the gravity gradient stabilized configuration the choice of α and ϵ combinations for the solar cell covered portion is limited. This condition practically eliminates the possibility of thermal storage temperature control for the solar cell covered portions of the vehicle. The solar cells considered for the SMS have an absorptivity of 0.78. The minimum weight of heat storage material needed is 10.15 lb/ft^2 for a flat surface and 6.46 lb/ft^2 for a cylindrical surface as shown by Figures 2-11 and 2-12 of Section 2. Heat storage material becomes prohibitively heavy. For example, the medium performance configuration with a solar cell area of 100 ft^2 requires 646 lb of heat storage material.

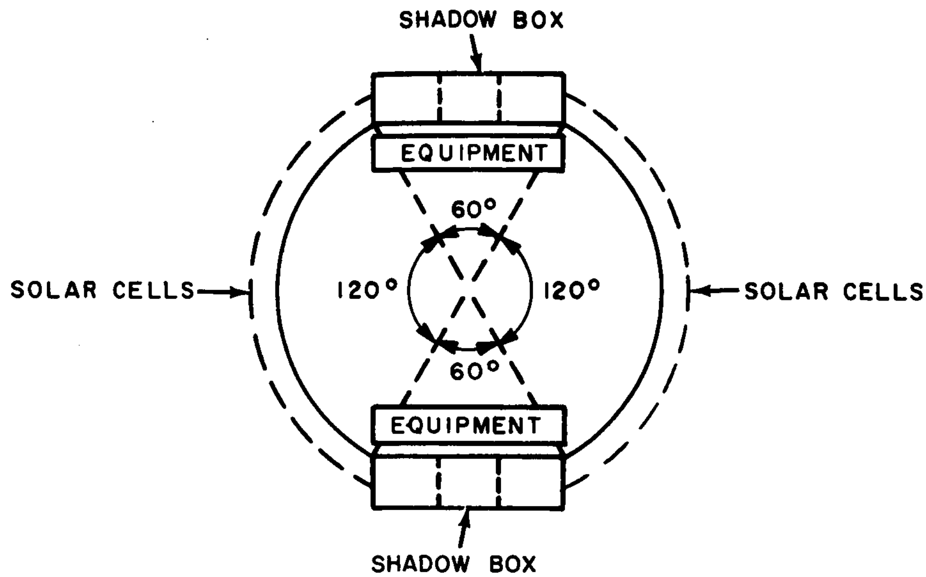


Figure 5-14. Solar Cell Distribution for Gravity Gradient Satellite

The temperature profile on the solar cells is shown in Figure 5-15. It may be seen that between 0° and 60° the cell temperatures are between 240°F and 125°F. These temperatures are considered too high for efficient cell operation. A semi-active fluid circulation loop weighing approximately 16 lb circulating a glycol water (or similar) fluid from the sunlit portion of the cells to the shaded portion is a possible solution for temperature control. Such a cooling scheme, in conjunction with the shadow box technique for equipment cooling, would be the lightest thermal control system for the gravity gradient stabilized spacecraft. It should be pointed out that a forced circulation system alone (without shadow boxes) can be used for thermal control of the gravity gradient vehicle. It is estimated that such a system would weigh approximately 25 lb for the medium performance vehicle. See Section 4 of Volume 5 for a discussion on forced circulation thermal control. The average spacecraft temperature history as a function of internal power density is shown in Figure 5-16.

H. PROBLEM AREAS

The gravity gradient satellite has a set of problems that are unique to its configuration. These may be divided into two general classes involving its stability and orientation, and physical structure. A simple demarcation does not exist since the physical characteristics are closely related to factors affecting stability and control. A summary of the problems that must be solved follows:

1. Stability and Control

(1) The means must be found for initial rough Earth orientation prior to erection of the satellite.

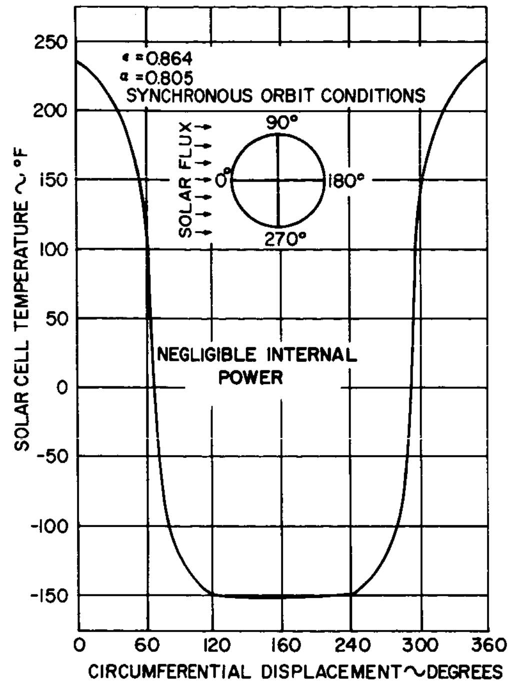


Figure 5-15. Solar Cell Temperature Distribution on a Cylindrical Satellite

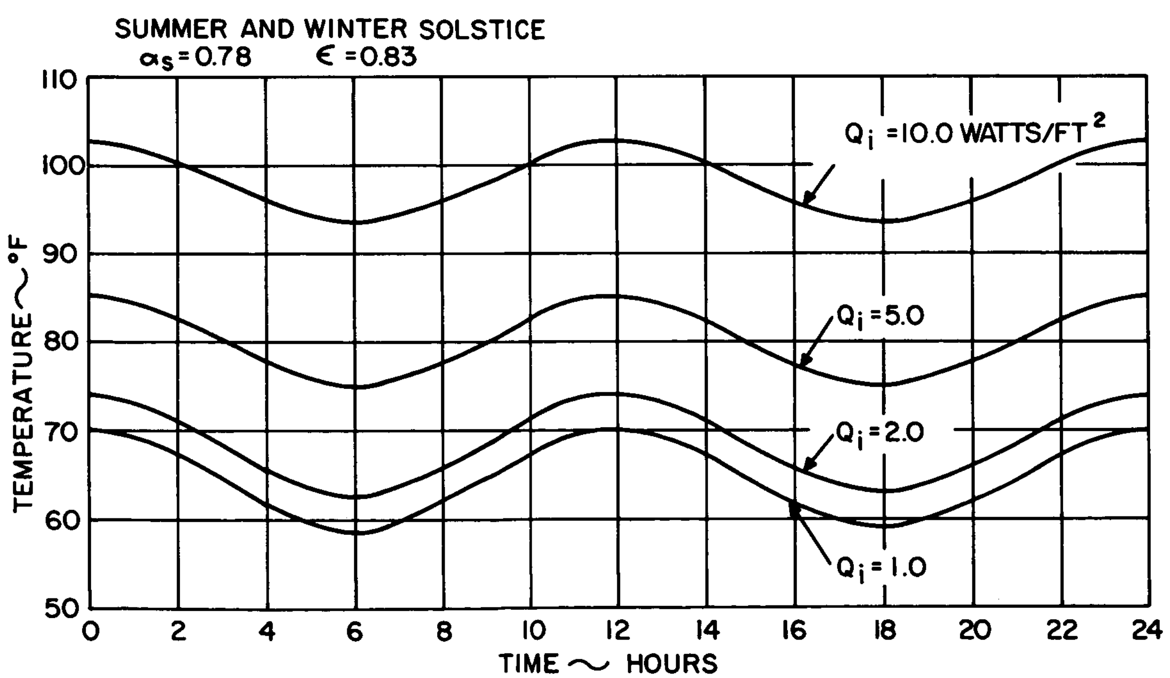


Figure 5-16. Variation of Average Surface Temperature

(2) The dynamic response of the satellite requires more consideration. The relatively light interconnecting structure between two massive sections may lead to uncertain dynamic response to corrective control torques.

(3) Passive means to control oscillation rates should be thoroughly investigated.

2. Structure

(1) A considerable amount of development work must be done to prove the feasibility of the proposed interconnecting structure.

(2) The thermal distortion problem must be investigated to determine the expected limits of misalignment and means of maintaining these limits. Correct alignment is imperative because it determines sensor positioning and station keeping jet orientation.

(3) Means of controlling and damping the two satellite sections during the erection cycle must be found.

SECTION 6 - SPIN STABILIZED SPACECRAFT

A. INTRODUCTION

Spin stabilization is a relatively simple manner of maintaining alignment of a satellite axis in inertial space. The basic requirement of dynamic balancing results in a symmetrical vehicle having several attractive features. The symmetrical shape minimizes the effect of unbalanced torque due to solar flux that tends to disturb axis alignment. At synchronous altitude, the disturbance due to solar pressure is sinusoidal in nature, and balances out over the course of a day. Further the gyroscopic effect of the spinning vehicle resists forces that tend to disturb its alignment.

The alignment of the spin axis normal to the equatorial plane, roughly parallel to the north-south Earth axis, provides an automatic semi-active thermal control. The spin motion rapidly presents the same surface to space and Sun, resulting in a relatively constant surface temperature. The mean surface temperature can be adjusted to the desired level by the use of a properly chosen surface coating. The temperature of internal components is regulated by conducting heat to or from the external surfaces. Common to all the satellites in synchronous orbit, the occult period presents a special problem.

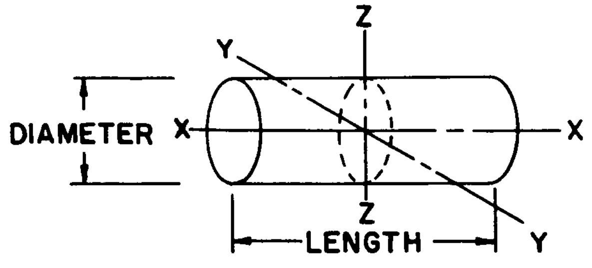
The spin motion makes an oriented solar array impractical, since the array would have to be de-spun to orient. Body mounted solar cells seem to be the only logical solution. In effect, they present an oriented array to receive solar flux although approximately three times the number of cells are required. The size of the satellite is largely determined by the solar power required. Systems requiring high power may require additional deployable panels if the space within the booster nose cone does not permit sufficient surface area to be developed on the basic satellite.

Methods of attaining the proper orbit and the sensing systems associated with these methods cause problems that tend to complicate the satellite. The most difficult problem of all is the effect of spin motion on sensor equipment. While the spin axis may remain practically motionless with respect to the Earth, the equipment is revolving at a considerable rate. In order to "stop" the Earth, some form of image motion compensation is required if the sensors are to work properly.

For this discussion, the concepts of the spacecraft presented are based on assumed power requirements and are limited to those that can be enclosed within the nose cone envelopes of the Thor-Delta (100 lb satellite), Atlas-Agena (500 lb satellite), and Atlas-Centaur (1000 lb satellite). Table 6-1 is a tabulation of the assumptions used for the models.

All satellites were in the form of hollow drums covered by solar cells. Equipment and sensors were located in a structural toroid in the center of the drum; weight was thereby controlled to maintain the proper realtionship of roll to pitch inertias (1.2 minimum). It was necessary to supply extra weight on external arms to the 500 lb vehicle to compensate for the unfavorable weight distribution caused by nose cone configuration. The Agena has a high length to diameter ratio.

TABLE 6-1
ASSUMPTIONS FOR SPIN STABILIZED SPACECRAFT



Launch Vehicle	Thor-Delta	Atlas-Agena	Atlas-Centaur
Satellite Weight	100 lb	500 lb	1000 lb
Power Estimate	83 W	230 W	350 W
Diameter (in.)	42	50	90
Length (in.)	36	84	72
I_{xx} - Roll	13.8 slug-ft^2	68 slug-ft^2*	346 slug-ft^2
I_{yy}, I_{zz} Pitch, 1/AW	9 slug-ft^2	57 slug-ft^2	205 slug-ft^2
Roll/Pitch Ratio	1.53	1.2	1.69
Structural Weight	35	97.0	151
Solar Cells	23	66	100
Wiring	6	33	65
Subsystem Weights	36	304	684
Adapter	3	20	40
Apogee Motor	78	750	1450

* Basic I_{xx} = 57 slug-ft^2. External flyweights required to attain 68 slug-ft^2 for minimum R/P Ratio = 1.2

A discussion of the various subsystems for the spin stabilized satellites follows. No detailed equipment studies are made, and all information on physical characteristics, such as weight and inertia, is based on statistical curves and data that can be found in Volume 6.

B. ANALYSIS OF A SPINNING VEHICLE

Fundamentally, the control concept of a spinning vehicle depends on the principle of nutation of a gyroscopic element when subjected to an impulsive torque. Figure 6-1 is a vector diagram of a complete control cycle. Initially, the spacecraft is spinning with an angular momentum $\vec{H}_S$ with its spin axis directed toward P_1. An impulse of momentum $\delta \vec{H}$ is applied at this time. The impulse of momentum $\delta \vec{H}$ is directed normal to the spacecraft spin axis and is contained in the plane defined by the central body, the cg, and the initial spin axis. Following the application of $\delta \vec{H}$, the total momentum is H_t and the spacecraft spin axis precesses about this total momentum vector at an angular velocity of ω_t where

$$\vec{\omega}_t = \vec{\omega}_s \left(\frac{I_s}{I_o}\right) \qquad (6\text{-}1)$$

where	ω	=	spin angular velocity
	I_S	=	spin inertia
	I_o	=	inertia about other two body axes, taken to be equal

When one spin axis again lies in the plane of $\vec{H}_S$ and $\delta\vec{H}$ directed toward point P_2, a second impulse is applied which causes nutation to stop. Thus, a net angular displacement of $2\,\Delta$ of the vehicle spin axis has been achieved.

The time to precess π radians from P_1 to P_2 is π/ω_t sec. During this time, the body has rotated through an angle equal to $\frac{\omega_t}{\omega_s}\pi$ radians, which is equivalent to $\frac{I_s}{I_o}\pi$ radians when Eq 6-1 is substituted. This implies that the two torque producing devices are to be separated on the body by an angle equal to $\frac{I_s \pi}{I_o}$ radians in order to accomplish a $2\,\Delta$ angular rotation of the spin axis in one desired direction without any residual precesion error.

The control system thus operates in the following manner. A trigger pulse is obtained from the fan shaped sensor as it sweeps by the edge of the central body. This pulse triggers the first torque device. When the body has precessed through an angle of 180°, the second torque is triggered on the basis of a computed time interval equal to π/ω_t sec. Figure 6-2 depicits the geometrical arrangement of the two torque devices and the triggering sensor device.

An alternate technique for spin control using a pinch plasma engine as a torque device is possible. This technique results in essentially the same performance characteristics and the same total impulse requirement.

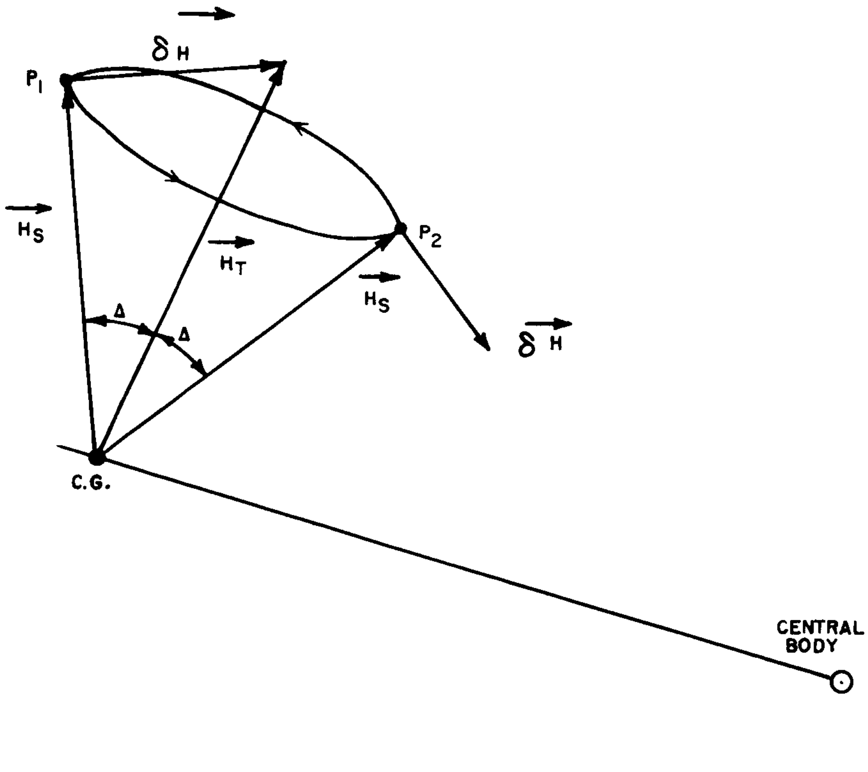
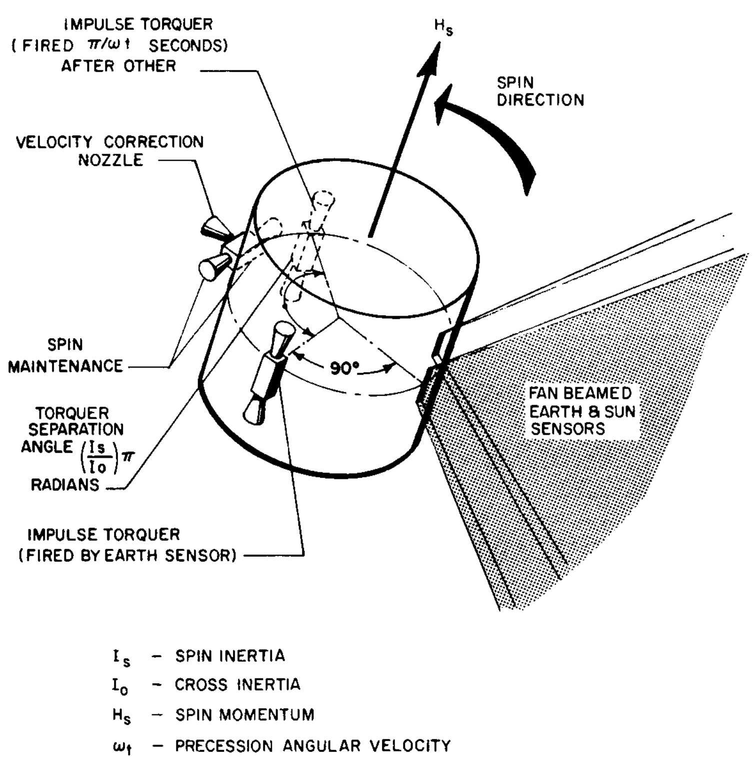


Figure 6-1. Spin Vehicle Momentum Vector Diagram



I_s – SPIN INERTIA

I_o – CROSS INERTIA

H_s – SPIN MOMENTUM

ω_t – PRECESSION ANGULAR VELOCITY

Figure 6-2. Geometrical Arrangement of Basic Spin Control Components

The 100 lb spacecraft has moments of inertia for I_S = 14 slug-ft^2 and I_O= 9 slug-ft^2. For a half cone angle Δ of 0.1°, $\delta \vec{H}$ = 0.2 in.-lb-sec for $\delta H = \Delta H_S$, when $H_S = I_s \omega_S$ and ω_S = 10 RPM. The impulse torque magnitude for a 50 millisec pulse would be 3.96 in.-lb. For a control lever arm of 19 in., a thrust level of about 0.2 lb is required.

In order to accomplish the SMS mission, the RMS rates to which the meteorological sensors may be subjected to are about 0.001°/sec maximum for a nominal performance level, with lower rates desired for improved resolution. If the SMS were to be spun at only 1 RPM (about 6°/sec) an image motion compensation (IMC) scheme with a basic accuracy of approximately 1 part in 60,000 would have to be evolved. The development of such a system which would be both light in weight and reliable appears to be a major task.

C. ATTITUDE CONTROL

The attitude control for a spin stabilized vehicle has been discussed in Section 4, Minimum Capability Spacecraft. The basic system shown in Figure 4-8 applies to any spinning configuration.

D. SENSORS AND COMMUNICATIONS

All of the sensor equipment proposed in the SMS studies has required a high degree of platform stability to properly perform its task. In addition, the transmission of the acquired data has been predicated on use of directed antennas of fairly narrow beamwidth. A rapidly rotating platform, oriented as the SMS satellite is, presents a different set of problems. Since the difficulty lies in de-spinning the motion in relation to the sensor imaging tubes, only this aspect will be discussed. Basic sensors data link and communications equipment remain essentially the same as previously described.

1. Visual Sensors

Image motion factors are introduced when spin stabilized satellites are employed as sensor platforms. For certain sensors such as the heat budget measurment sensor, the spin of the vehicle could be utilized to replace an otherwise necessary scanning function. Cloud cover sensors operating in the visible spectrum employ image tubes such as a vidicon or image orthicon. These sensor tubes, like a camera film, will record any object movement occurring during the exposure interval. Since the vehicle platform rotation axis is parallel to the Earth, the sensor, in effect, sweeps across the Earth at a rate comensurate with the spin rate. This sweep action causes a ground image smear in proportion to the exposure time and consequently, degradation of resolution.

Since relatively long exposure times are desirable, particularly for low light level imaging, a means must be employed to de-spin the sensor and hold the image stationary on the photocathode to within the limits necessary to reduce ground smear to one-half of the resolution element.

The problems associated with employing a de-rotation mirror are formidable. It would be necessary for the attitude control system to supply signals to control the rate of de-rotation to an accuracy of 0.001°/sec. Precision mirror drive systems cause further control degradation. For these reasons, the implementation of this approach would be marginal if at all feasible.

Stop motion techniques employing fast shutter speeds to reduce ground smear to tolerable limits limit the sensor exposure time to snapshot operation commensurate with the satellite spin rate and allowable ground smear. For example, a 10 RPM spin rate would require a maximum exposure time of 0.0002 sec to maintain a 10 mi resolution. This shutter exposure time limits surveillance to times of high scene brightnesses.

As previously stated, longer exposures are desirable for maximum utilization of sensor capabilities, especially for low light level imaging. If control systems were provided, the image motion compensation could be performed by means of a de-rotation mirror to extend the exposure interval to better advantage than fast shutter speed.

2. Heat Budget System

In order to incorporate a high resolution heat budget system in a spinning vehicle, it is desirable that the spin axis be used as one of the scan parameters to avoid the necessity of de-spinning. The frame time in seconds will be $t = \frac{3.33}{RPM}$ with the scanning axis parallel to the spin axis. In order to utilize practical time constants for the detector, the resolution must be limited. For 1 millisec time constant, the resolution element is equal to approximately 500 x (RPM)1/2 miles on a side. A vehicle spinning at 4 RPM would have a 1000 mi resolution element.

A quantum detector might improve this resolution by a factor of 30 but other restrictions (limited spectral response and necessity for active cooling systems) negate its use.

3. Communications

Since the SMS is spinning, the sensor data antenna must be made omnidirectional in the direction of the Earth. Its pattern in the plane containing the axis must have a beamwidth of greater than 17° to illuminate the Earth. A suitable antenna would be a slotted dipole array having an omnidirectional "pancake" beam. Its gain would be approximately 6 db as compared to 19 db for the parabolic antenna that is used in the medium capability satellite. An alternative system would use a directional antenna system with an electronically or mechanically switched beam. This would have a gain of approximately 17 db. However, the increased weight and complexity of this antenna might well overcome the increase in gain obtained. The net result is the necessity for more power in the vehicle, or loss of safety margin.

E. POWER SUPPLY

The power supply of the spin stabilized satellite presents few problems, since its alignment in synchronous equatorial orbit tends to present an oriented array to the Sun. Due to the inclination of the Earth orbit, the Sun angle varies

±23-1/2° from the normal during the year. This factor must be taken into account when determining the required area.

The total area of solar cells, however, is over three times as great as that of an oriented panel, due to the fact that the cells are completely distributed about the body and only one third of them can be counted on to generate power at any instant. This factor imposes an undesirable weight penalty on a spinning vehicle that requires appreciable power. Figure 4-11 indicated the trend of specific weights of power arrays as the total required area increases. A complete discussion of this type of power supply will be found in Section 3 of Volume 5.

F. THERMAL CONTROL

Generally there are three types of temperature control systems associated with spacecraft (they are given in the order of increasing complexity and ability to maintain temperatures within narrow limits): 1) passive, 2) semiactive, and 3) active (see Section 4 of Volume 5 for a definition of these terms). The spin stabilized configuration is most suited for a passive thermal control system, as it is an intrinsic semiactive system by virtue of the spin.

The histories of the solar heat inputs to the three basic satellite configurations during their entire life (one year) are shown in Figure 6-3. As shown in Section 4 of Volume 5, the only significant external heat input to a satellite at synchronous altitudes is the solar flux. The solar heat impinging on the satellite at any time of the year is given by

$$Q_s = \alpha S \left[LD \sin \phi + \frac{\pi D^2}{4} \cos \phi \right] \qquad (6\text{-}2)$$

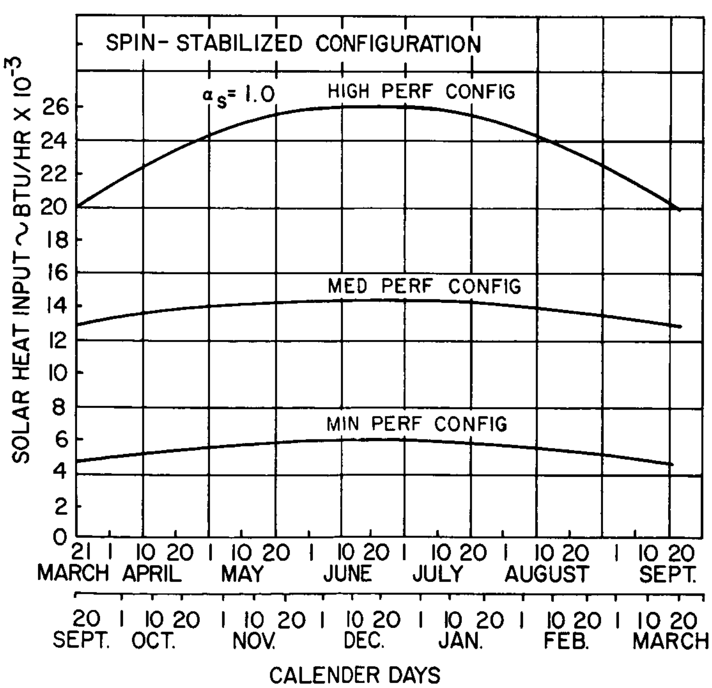


Figure 6-3. Variation of Solar Heat Input with Calendar Time

where Q_S = solar heat, Btu/hr

α_S = absorptivity of surface for solar radiation

S = solar constant, 442 Btu/hr-ft^2

L = satellite length, ft

D = satellite diameter, ft

ϕ = angle between the Sun-Earth vector and the satellite axis of rotation

The angle ϕ is given by

$$\phi = \cos^{-1}\left[\operatorname{Sin}\Delta \operatorname{Sin}\omega_{E/S} t\right] \tag{6-3}$$

where Δ = angle between the Earth axis of rotation and the normal to ecliptic plane - 23.5°

$\omega_{E/s}$ = angular velocity of the Earth around the Sun, radians/day

t = time from autumnal Equinox, days

Equating the external heat (as given by Eq 6-2) and the internal power dissipation Q'_i to the radiated heat, the following expression is obtained for the average spacecraft temperature

$$T = \left[\frac{\alpha S\left(LD \operatorname{Sin}\phi + \frac{\pi D^2}{4} \operatorname{Cos}\phi\right) + 3.413\, Q'_i}{\sigma \epsilon D\left(\pi L + \frac{\pi D}{2}\right)}\right]^{1/4} \tag{6-4}$$

where σ = Stefan-Boltzmann constant, Btu/hr/ft^2/°R^4

ϵ = emissivity of the material

Figure 6-4 presents the temperature histories of the minimum, medium, and the maximum performance spin stabilized configurations.

The local temperature distribution on the circumference of the cylindrical satellite is shown in Figure 6-5 for various RPM's. It can be seen that even moderate rotational speeds prevent excessive temperature excursions. The individual component temperatures are not expected to vary by more than a maximum of 10°F from the temperatures shown in Figure 6-4. Since the average power density of any of the three spin stabilized configurations is of the order of only approximately 3 W/ft^2, a temperature difference greater than 10°F can be prevented with little weight penalty for the inclusion of passive heat conduction paths.

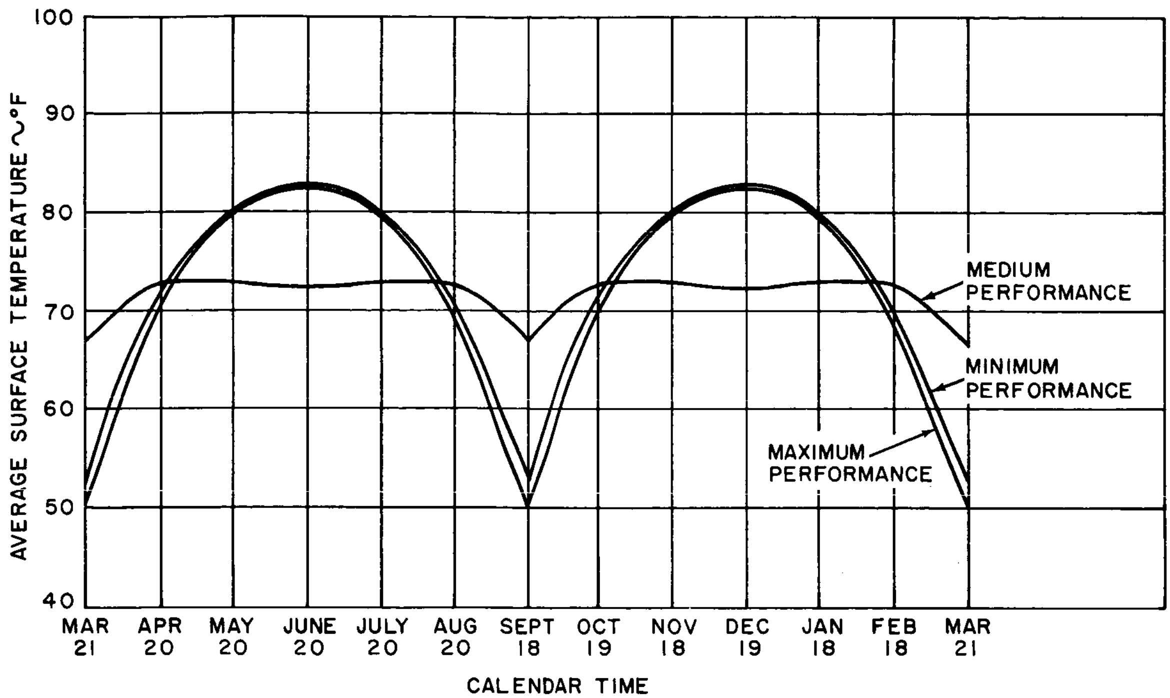


Figure 6-4. Temperature Distribution on a Spin Stabilized Cylindrical Vehicle

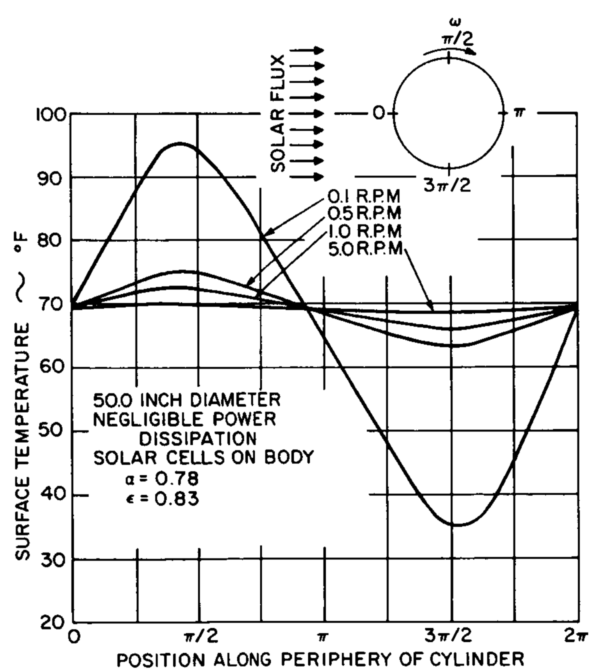


Figure 6-5. Temperature Distribution vs RPM

The temperature of the spacecraft during the occult periods is an important consideration. The analysis is presented in Section 4 of Volume 5. Figures 4-23, 4-24, and 4-25 of that section show the spacecraft structural temperatures for the minimum, medium, and maximum performance configurations. During the occult period, power will be supplied for surface heating elements to maintain the temperatures of critical components at a safe level. (See Section 4 of Volume 5 for a discussion of surface heating elements.)

G. STRUCTURE

Figure 4-1, which shows the arrangement of the minimum capability satellite, serves to illustrate some of the inherent problems in spinning satellites. The structural arrangement consists of a basic central torodial compartment in which most of the equipment is housed. Attached to each end of the toriod are circular skirts supporting the solar cell modules. Enough secondary structure is supplied to support any auxiliary equipment not housed in the central toroid.

Apogee motor loads are directly reacted in the central ring. Loads imparted during launch are also carried, through an intermediate adapter, to this central ring. The load paths described were chosen to isolate the solar cells and their light substrate from any loads other than those imparted by their own weight.

A factor which may be decisive in determining certain aspects of the configuration is the choice between retaining or jettisoning the apogee kick motor after the synchronous orbit has been achieved. If it is assumed that the image motion compensation problem has been solved, then the platform stability again becomes a critical factor in mission accomplishment. Figures 4-3 and 4-4 indicate the effect of unbalance on "wobble" of the spin axis. The wobble can be induced by the mislocation of the final cg of the burned out apogee motor which cannot be predicted with the degree of precision necessary. If the motor is not jettisoned, thereby returning the spacecraft to a known and tested balance condition, then some rather sensitive dynamic balancing mechanism must be incorporated in the spacecraft to cancel any unbalance caused by the burned out apogee motor.

H. PROBLEM AREAS

The primary problem associated with the spin stabilized satellite concerns the mechanism for image compensation. Vehicle rotation must be accurately determined and the image de-spin mechanism precisely synchronized.

Other problems previously discussed in Section 4 can be defined as:

(1) Development of sensors with adequate unattended life.

(2) Development of mechanism to counteract wobble of the spin axis due to damage or equipment shift.

(3) Development of means to prevent direct impingement of the Sun's rays on sensitive detectors.

(4) Means of acquiring the correct star (Polaris) with the star tracker.

(5) Development of horizon scanners with the capabilities required for this mission.

SECTION 7 - RELIABILITY

A. INTRODUCTION

The reliability studies conducted under this design study contract for the Synchronous Meteorological Satellite (SMS) were directed toward determining sets of system parameters which would result in various fixed levels of satellite reliability within the range of 0.50 to 0.99 for a one year period of operation. This particular range was chosen because it was believed to be the most practical for illustrating absolute minimum requirements and the merits of various alternative approaches and reliability trade-offs. The choice of 0.50 as the minimum probability of success was made for the purpose of confining the investigation within an area of reasonable feasibility. Republic does not contend that such a low probability of success should be acceptable, but on the contrary, uses this value to emphasize the necessity of devoting considerable attention to reliability in design. The upper limit of 0.99 probability of success was chosen because it was believed that a higher level of reliability would be extremely difficult, if not impossible, to attain within the weight limitations and the forseeable state of the art.

In these studies, the SMS satellite was divided into its major functional subsystems to facilitate reliability analysis on a parametric basis.

As illustrated in Figure 7-1, the reliability of the overall SMS system may be expressed in terms of the reliability parameters of its seven major subsystems, namely:

(1) Structure

(2) Thermal Control

(3) Power Supply

(4) Stabilization and Control

(5) Data Handling

(6) Communications

(7) Sensors

In addition to the foregoing satellite subsystems, there are ground station facilities which must be integrated with satellite counterparts. For the purposes of this reliability study, failures within ground station equipment will not be considered in the prediction of SMS reliability, because the combined operational effectivity of a satellite-ground station complex is more properly a matter to be considered in an operations analysis at a later phase of the program. However, the problems of satellite-ground station compatability will be treated as interface requirements imposed upon the satellite system.

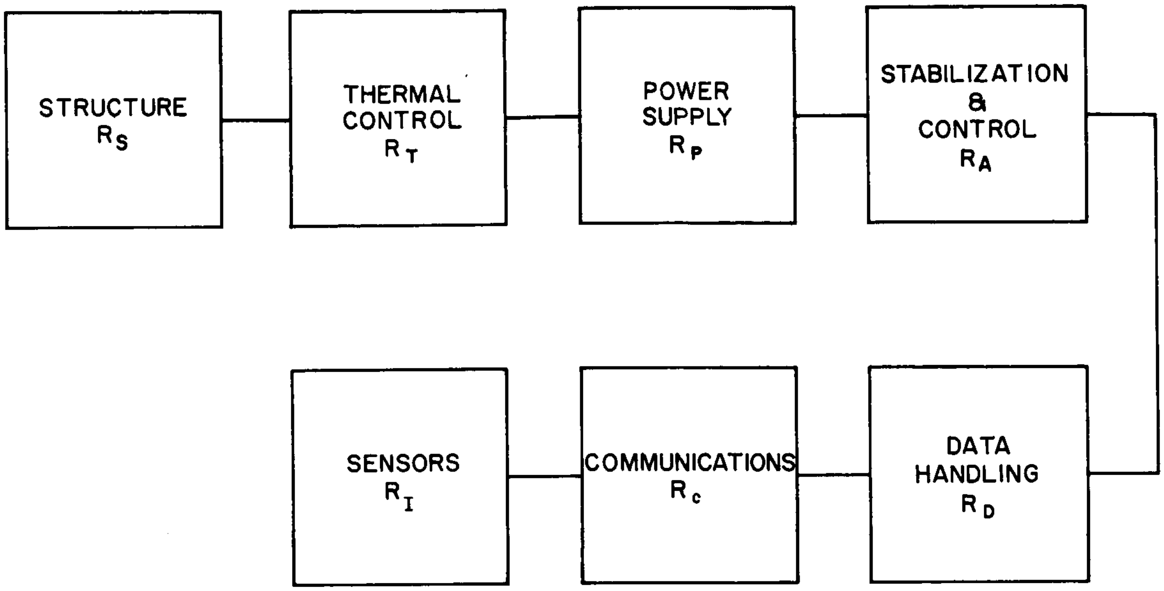


Figure 7-1. System Reliability Model

The reliability of the overall system and individual subsystems and components was assumed to be governed by the exponential failure rate law. The quantitative results expressed throughout this study were computed on the basis of the well known formula, $R(t) = e^{-\lambda t}$ as applied in standard reliability engineering practice. This formula is correct for all properly debugged devices which are not subject to early failures (infant mortality) and which have not suffered any degree of wearout damage or performance degradation because of age. The life period which is validly covered by this formula is conventionally referred to as the useful life of the device. For the SMS mission, the maximum value of the time, t, will be 8760 hours, corresponding to one year. In the selection of components for the SMS, it is important that the useful life of each component be known, and be greater than 8760 hours. This restriction is dictated by the practical aspects of system reliability, and is not to be considered as an attempt to constrain the system design of this particular formula.

Two basic parameters must be considered in reliability engineering, namely the useful life and the chance failure rate.

For a reliabile system, it is a prerequisite that it shall not be expected to perform in the region of time where there is any significant effect of wearout failures. Thus, a reliable system must operate only within the chance failure region, as shown in Figure 7-2.

In selecting components for, and in the design of a prototype SMS system, caution should be excerised in evaluating mean time before failure (MTBF) and useful life figures quoted by prospective suppliers. An item may have an MTBF rating many times greater than its useful life, and conversely, the useful life may be much greater than the MTBF rating. The MTBF rating is determined by dividing the accumulated operating hours on several units which have been tested, usually concurrently, by the number of failures experienced. Thus, for example, 500 units,

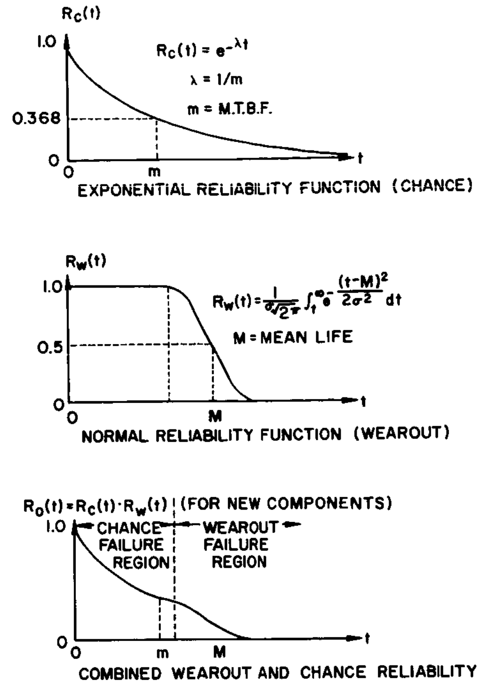


Figure 7-2. Combined Wearout and Chance Reliability

each being subjected to 100 hours of testing, experience 10 failures in the entire lot. The MTBF of 5000 hours is computed by dividing the 50,000 cumulative hours by the 10 failures experienced. Although the 5000 hour MTBF rating indicates that the components will have a relatively low probability of chance failure, it should not be interpreted as a guarantee of useful life. As shown in Figure 7-2, the probability of an item failing up to any given time may be expressed as the sum of the probabilities that it may fail through chance or wearout. The probability of wearout failure is essentially zero until the wearout time region is approached. The probability distribution for wearout failure is substantially a normal distribution centered about the mean wearout time. The probability of failure is $[1 - R(t)]$ which for chance failures reduces to $[1 - e^{-\lambda t}]$, an exponential function. The exponential function for a 5000 hour MTBF has a rather slow rise, but as the wearout region is approached, the sun of the chance and wearout failure probabilities rises very rapidly. Thus, to ensure a reliable item, the wearout region must be located sufficiently distant in time from the mission termination time so that the total failure probability at the termination of the mission is held to a value corresponding to the required level of reliability. Therefore, both the chance (MTBF) and the wearout (useful life) characteristics of each component and subsystem must be considered.

In this study, where a failure rate or MTBF is quoted in the quantitative data, there has been an implicit assumption that the useful life is sufficiently high so that there is negligible effect of wearout failure probability.

B. MISSION REQUIREMENTS AND FAILURE CRITERIA

1. General

Before a complete meaningful statement can be made as to the reliability of a complex functional system such as a meteorological data satellite, a precise definition of adequate performance must be provided. For the SMS system, the mission requirements will define the basic performance requirements. But further specification is required to define the minimum acceptable quantity and quality of meteorological data, so that standard reliability techniques can be properly applied. One the description of adequate performance has been established, the definitions of success and failure follow automatically, failure corresponding to inadequate performance.

A failure (or malfunction as it is sometimes called) is the cessation of an item's ability to perform its specified function within the tolerance limits specified for any portion of time, or for any cycle during which the function is required. Failure may therefore be divided into two general classes:

(1) Inadequate operation - this includes failure to operate, as well as operation outside tolerance limits.

(2) Untimely operation - operation that is out of phase with the functional requirements.

Failure of any item may in turn cause inadequate or untimely operation of other items in a complex system. The failures caused in this way are referred to as secondary failures.

As prescribed by the NASA work statement, the SMS system shall have a reliability adequate for one year of orbital operation, in which a 24 hour daily surveillance capability is provided, and cloud cover data is obtainable at least once every thirty minutes.

For purposes of the reliability study, it will be assumed that the data relay function will not interfere with, or degrade the reliability of the primary meteorological data gathering function, although meteorological data transmission will be temporarily interrupted during times when data is being relayed. Thus, it will be assumed that the data relay capability will be available at all times when the primary mission capability is present. Such as assumption is reasonable, in that the only equipment used for the data relay which is not directly connected with the primary mission will be wideband receiver. However, this receiver will not be used solely for the data relay, but will also be used for range and range rate measurements during the ascent phase of the primary mission.

2. Failure Criteria for Meteorological Data Coverage

Certain problems are presented in establishing the failure criteria to be used with regard to the primary SMS mission of obtaining and communicating meteorological data to designated Earth stations. The ultimate criterion for primary mission success will be the value of the data to the meteorologist. The ability of a trained observer to decipher data of varying degrees of quality is difficult to predict.

In general, it will vary with different individuals. Too literal an application of the standard definition of failure to cases where data somewhat less than perfect, but still usable, were recovered, would unduly penalize the system and result in an unrealistically pessimistic appraisal of its reliability. Therefore, until such time as a better appraisal of the value of specific data quality levels is available, the standard of performance for the SMS system should be set at that level, where the data received can be processed and reconstructed into a useful image. As to the frequency at which observations of the Earth's disc can be made, the standard of performance of one picture every thirty minutes, as prescribed by the NASA work statement, shall govern.

3. Failure Criteria for Subsystems and Components

As to the subsystems and components which will make up the SMS system, the application of the standard criteria for failure should not present too many problems. In general, a subsystem or component has failed when its ability to perform its specified function ceases. This definition includes the failure modes wherein the performance rendered is outside of the specified tolerance limits, as well as the catastrophic failure modes.

In the design of a prototype SMS system, caution will have to be exercised in setting the tolerance limits on individual subsystems to avoid the situation in which overall system performance is inadequate, although all subsystems are operating within the tolerance limits. Such a condition can arise through the interaction of the combined subsystems with adverse tolerances.

C. ENVIRONMENTAL FACTORS

A proper selection of parts and materials (made on the basis of the best available data pertaining to the effects of the operational environments which will be experienced) is a prerequisite to attaining satisfactory reliability in any spacecraft.

These data can be obtained either through the results of experiments in which actual material samples were subjected to flight through a space environment or from the results of laboratory tests with simulated space environments.

Both types of data can be obtained from government publications, technical journals, and private testing organizations. The table of environmental characteristics and their effects upon materials and parts, which was prepared for this report, was compiled from a survey of such literature.

This environmental data is needed because, to assure a reliable design, it is necessary to derate spacecraft materials and components so that they will be subjected to sufficiently low stress levels. These stress levels determine the failure rates and the useful life, and hence, reliability.

The term stresses as applied to failure phenomena has a generic meaning, and includes all physical factors which have an adverse effect on continued life and desired performance. These factors are not limited to mechanical stresses, but include other environmental parameters such as temperature, ambient pressure, vibration, and radiation.

As in conventional mechanical design practice, a system must be designed for satisfactory performance under the most severe combination of stresses. For the SMS, the stress combinations which will be experienced in each of the three basic phases of operation must be examined to determine the worst combination for each subsystem and the overall system. These three phases are:

(1) Ground handling

(2) Orbit attainment (boost, parking, and transfer)

(3) Orbit

The SMS will experience a variety of environments in ascending to, and operating in its circular synchronous orbit 22,400 miles above the Earth's surface. Prior to launch (on the ground) the satellite will be subjected to the environments of manufacture, storage, transportation, and prelaunch checkout. During ascent, it will encounter the various layers of the Earth's atmosphere, and the space environments of the parking and transfer orbits, in addition to the dynamic conditions associated with launch and transferring from one orbit to another.

For the region in space beyond 100 miles above the Earth's surface, including the synchronous orbit, the satellite environment will include:

(1) High vacuum

(2) Magnetic fields

(3) Gravitational fields

(4) Micrometeorities

(5) Cosmic rays

(6) Electromagnetic radiation including
  a. Ultraviolet rays
  b. X-rays
  c. Gamma rays

(7) Neutrons

(8) Charged electron and proton particles

Table 7-1 summarizes the effects of the aforementioned environmental factors upon spacecraft equipment and materials, and indicates what remedial measures should be adopted in the system design.

Table 7-1 is qualitative rather than quantitative in nature, and should be used as a guide in the design of the SMS. Unfortunately, such a table does not provide a means of estimating the amount of derating required to achieve a specified failure rate for a given component or system. The determination of appropriate derating factors must be left to the design phase of a specific system becuase it is necessary to know the exact types of components and their strength ratings. Component strength not only means its mechanical resistance to vibration, shock, pressure, or acceleration, it also includes thermal strength as resistance against environmental and self-generated temperature and against

TABLE 7-1
EFFECTS OF SPACE ENVIRONMENT

Environment	Effects	Design Factors
Temperature	Thermal energy within the vehicle produced by solar radiation, Earth shine, Earth radiation, and internal heating. No convection heating outside Earth atmosphere.	Design for temperature control by means of absorptive and reflective surfaced (α/ϵ ratio) with heat control servo circuits. Isolate internal equipment thermally for temperatures between 0 and 60°C depending on requirements. For heat transfer, use radiation and conduction heat sinks. Spin space vehicle to eliminate temperature gradients around surface.
Vibration	Unimportant in space (except for launch environment). No acoustic, frictional, or combustion vibration problems except for special applications.	Any vibration, acceleration, or shock levels in space which may occur would be very small compared with those during the boost phase. The equipment must be designed to withstand the levels during boost, and, therefore, the lower levels encountered in space should pose no problems.
Acceleration	Unimportant in space except for special applications.	
Shock	Unimportant in space except for meteorite and micrometeorite impacts.	
High Vacuum	Sublimation and evaporation of materials occur in high vacuum.	Use materials with low sublimation rates. Allow sufficient thickness for sublimation and evaporation over expected operating life.
	Chemical atmosphere produced by outgassing and sublimation may have corrosive, plating, or chemical effects.	Select material with care to avoid hazardous conditions.
	Electrical arc-over or corona discharge.	Provide adequate insulation material and insulation paths.
Magnetic Fields	No effects except for fine instrumentation.	Avoid use of instruments not shielded against variations outside Earth's magnetic fields.
Gravitational Fields	No effects on materials or parts.	None for materials or parts. For manned vehicles, physiological considerations are involved.

TABLE 7-1 (Cont'd.)

Meteorites and Micrometeorites	Collisions with particles of varying sizes occur.	Statistically calculated risk is involved. Use pre-roughened surfaces or oxide finishes to minimize changes in α / ϵ ratio. Use sufficient outer skin thickness or secondary outer shell to provide protection against small particles.
Ultraviolet Light	Increases sublimation rates in high vacuum.	Minimize sublimation by selection of materials and by providing sufficient material thickness allowances.
X-rays and Gamma Rays	Ionization of material occurs, possibly causing atomic displacements which produce changes in material characteristics or composition.	Intensity of primary radiation is negligible; but shielding with heavy material may be considered for secondary ionizing radiation effects.
Neutrons	Intensity too low to require consideration of atomic displacement effects.	No special precautions necessary because of low intensity.
Trapped Electrons	Ionizing radiation occurs primarily in the Van Allen belts, possibly causing atomic displacements which produce changes in material characteristics or composition.	Protection required for externally mounted equipment such as solar cells. Space vehicle shell normally provides protection for internal equipment.
Trapped Protons	Ionizing radiation occurs primarily in the Van Allen belts, possibly causing atomic displacements which produce changes in material characteristics or composition.	For low energy protons in the outer Van Allen belts, protection requirements are similar to those for trapped electrons. For high energy protons in the inner Van Allen belts, there is no known adequate protection.

temperature cycling, electrical strength, (which includes the capability to withstand electrical potentials), and changes of the potential (such as frequency, resistance against humidity, corrosion, radiation, etc.). While it is well known that a decrease in stress level results in a reduction of failure rate, and that an increase in stress levels produces a rapid increase in failure rate, the relationship between stress and failure rate is by no means linear. This is difficult to determine by purely analytical techniques. Establishing a generalized relation between stress parameters and component strength and failure rate is difficult because no known method exists to express the combined effect of the stress parameters in a single number.

The observable effects of component failure, in the time domain, as a function of stress level must be determined from statistical testing. The results of such testing should be available from potential suppliers. When used properly, they should provide a reasonable indication of the effects of deviations from the nominal rated stress level upon component failure rates.

D. SUBSYSTEM RELIABILITY REQUIREMENTS

1. Overall System Reliability Requirements

As shown in Figure 7-1, the reliability of the overall SMS system is equal to the product of the reliabilities of its major subsystems. The reliability of any device may be defined as the probability that it will render adequate performance for the period of time intended, under the operating conditions encountered. Sometimes, this probability is referred to as the probability of survival. However, mere survival is not adequate performance in the case of most systems, including the SMS. The precise definition of what constitutes adequate performance is most important in reliability studies (see subsection B).

The probability that a given device will render satisfactory performance at time t, measured from the beginning of the mission, is given by the exponential law of reliability; $R(t) = e^{-\lambda t}$, where λ is the chance, or random, failure rate. Where several statistically independent subsystems are required to function jointly, the probability that all will function satisfactorily is the product of their individual reliabilities. Using the exponential formula, the overall combined reliability may be expressed as $R_o(t) = e^{-\lambda_o t}$ where $\lambda_o = \sum_{i=1}^{n} \lambda i$, the λi's representing the failure rates of the individual subsystems, numbered 1 through n. In equating the overall failure rate to the sum of the individual failure rates, there is an implicit restriction that the operating time, t, be common to all subsystems, because it is treated as a common factor in the exponent.

This procedure is also applicable to components and piece parts within subsystems and modules, as well as to subsystems within an overall system. For convenience in estimating system, subsystem, component, and piece part reliabilities, Table 7-2, and Figures 7-3 and 7-4 have been presented. In this table, the exponential function e^{-x} is tabulated for values of x ranging from 0 to 1.0. This function corresponds to the reliability function, $R(t) = e^{-\lambda t}$, where the general parameter x corresponds to the λt in the exponent. For a single system, subsystem, or component, the product of the failure rate and operating time is directly equal to x. For multicomponent subsystems, and subsystem combinations, x represents the product of the sum of the individual failure rates and the common operating time, $\left(\sum_{i=1}^{n} \lambda_i t\right)$.

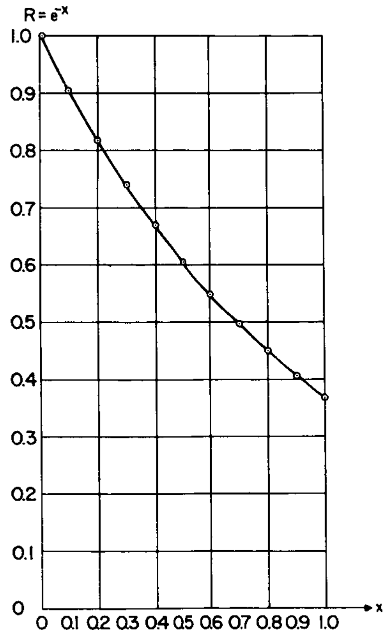


Figure 7-3. The Exponential Reliability Function - Range 0 to 1.0

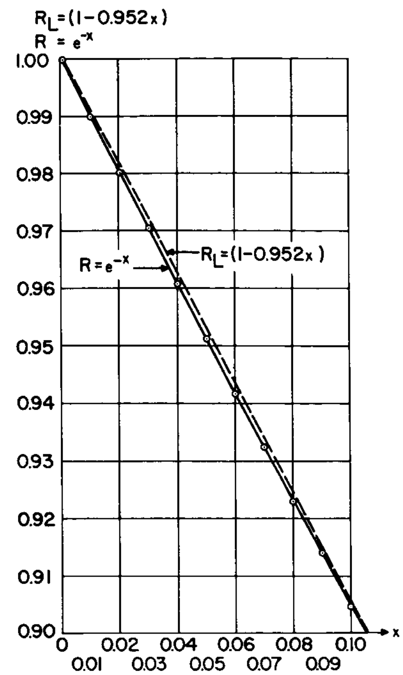


Figure 7-4. The Exponential Reliability Function - Range 0 to 0.1

To determine the total subsystem failure rates corresponding to given levels of overall system reliability, the value of x corresponding to the specified reliability is determined from either the graphs or the table. This value of x must be divided by the number of operating hours required (8760 hours for a one year mission) to obtain the failure rate. For example, a reliability of 0.99 for one year, corresponds to a value of 0.01 for the exponent, x. The failure rate, λ_o, is equal to 0.01 divided by 8760, or 1.142×10^{-6} failures per hour. The mean time before failure (MTBF) represented by m, is the reciprocal of the failure rate, or 876,000 hours per failure. Table 7-3 shows the relation between overall satellite reliability, total failure rate, and MTBF for a one year mission.

Consider the overall SMS system as composed of seven major subsystems, all of which must function properly for a successful mission. The sum of the seven individual failure rates determines the overall system failure rate, λ_o and, hence, the overall reliability. In the general case, the individual subsystem failure rates will differ from each other depending upon subsystem configuration and component failure rates. To get a first estimate of the levels of subsystem reliability required for the overall system reliability range of 0.50 to 0.99, it can be assumed that six of the subsystems each have the same failure rate, and the seventh, corresponding to the structure, has a zero failure rate, (Reliability = 1). Thus, the individual failure rates will be equal to one-sixth of the overall failure rate. In terms of the parameter $x = \lambda t$, the value of $x_o = \lambda_o t$ corresponding to the overall system is related to $x_i = \lambda_i t$ which represents the individual subsystems, according to $x_i = \frac{x_o}{6}$. Table 7-4 expresses the

TABLE 7-2
EXPONENTIAL FUNCTION, e^{-x}

x	e^{-x}	x	e^{-x}	x	e^{-x}	x	e^{-x}
0.000	1.0000	0.260	0.7711	0.520	0.5945	0.780	0.4584
0.010	0.9900	0.270	0.7634	0.530	0.5886	0.790	0.4538
0.020	0.9802	0.280	0.7558	0.540	0.5827	0.800	0.4493
0.030	0.9704	0.290	0.7483	0.550	0.5769	0.810	0.4449
0.040	0.9608	0.300	0.7408	0.560	0.5712	0.820	0.4404
0.050	0.9512	0.310	0.7334	0.570	0.5655	0.830	0.4360
0.060	0.9418	0.320	0.7261	0.580	0.5599	0.840	0.4317
0.070	0.9324	0.330	0.7189	0.590	0.5543	0.850	0.4274
0.080	0.9231	0.340	0.7118	0.600	0.5488	0.860	0.4232
0.090	0.9139	0.350	0.7047	0.610	0.5434	0.870	0.4190
0.100	0.9048	0.360	0.6977	0.620	0.5379	0.880	0.4148
0.110	0.8958	0.370	0.6907	0.630	0.5326	0.890	0.4107
0.120	0.8869	0.380	0.6839	0.640	0.5273	0.900	0.4066
0.130	0.8781	0.390	0.6771	0.650	0.5220	0.910	0.4025
0.140	0.8694	0.400	0.6703	0.660	0.5169	0.920	0.3985
0.150	0.8607	0.410	0.6637	0.670	0.5117	0.930	0.3946
0.160	0.8521	0.420	0.6570	0.680	0.5066	0.940	0.3906
0.170	0.8437	0.430	0.6505	0.690	0.5016	0.950	0.3867
0.180	0.8353	0.440	0.6440	0.700	0.4966	0.960	0.3829
0.190	0.8270	0.450	0.6376	0.710	0.4916	0.970	0.3791
0.200	0.8187	0.460	0.6313	0.720	0.4868	0.980	0.3753
0.210	0.8106	0.470	0.6250	0.730	0.4819	0.990	0.3716
0.220	0.8025	0.480	0.6188	0.740	0.4771	1.000	0.3679
0.230	0.7945	0.490	0.6126	0.750	0.4724		
0.240	0.7866	0.500	0.6065	0.760	0.4677		
0.250	0.7788	0.510	0.6005	0.770	0.4630		

TABLE 7-3
OVERALL SMS RELIABILITY vs TOTAL FAILURE RATE AND MTBF FOR A ONE YEAR MISSION
(8760 Hours)

Overall SMS Reliability	λ_o $\frac{\text{Failures}}{\text{Hour}}$	m = MTBF $\frac{\text{Hours}}{\text{Failure}}$
0.99	1.14×10^{-6}	876,000
0.95	5.71×10^{-6}	175,200
0.90	11.42×10^{-6}	87,600
0.85	18.27×10^{-6}	54,800
0.80	25.11×10^{-6}	39,900
0.75	33.11×10^{-6}	30,100
0.70	40.81×10^{-6}	24,300
0.65	49.09×10^{-6}	20,400
0.60	58.22×10^{-6}	17,150
0.50	78.77×10^{-6}	12,700

relations between overall system reliability and individual subsystem reliability for the case where $x_i = \frac{x_o}{6}$.

TABLE 7-4
RELATIONSHIP OF OVERALL SYSTEM TO SUBSYSTEM RELIABILITY FOR $x_i = \frac{x_o}{6}$

Overall SMS Reliability	x_o	x_i	Individual Subsystem Reliability	Individual Subsystem Average Failure Rate (Failure/Hour)
0.99	0.01	0.00167	0.9983	0.190×10^{-6}
0.95	0.05	0.00833	0.9915	0.952×10^{-6}
0.90	0.10	0.0167	0.9826	1.903×10^{-6}
0.85	0.16	0.0267	0.9733	3.045×10^{-6}
0.80	0.22	0.0367	0.9635	4.185×10^{-6}
0.75	0.29	0.0483	0.9532	5.518×10^{-6}
0.70	0.36	0.0600	0.9423	6.801×10^{-6}
0.65	0.43	0.0716	0.9306	8.181×10^{-6}
0.50	0.69	0.1150	0.8905	13.128×10^{-6}

Although the foregoing tabulation assumes a uniform level of reliability among six major subsystems (which may not conform exactly with the failure characteristics of the actual equipment) it does serve a useful purpose in illustrating the level of subsystem reliability and the average subsystem failure rate necessary to provide various specified levels of overall system reliability. A reliability of 0.8905 is required in each subsystem to produce even a 0.50 reliability in the overall system. For higher overall system reliability levels, the required subsystem reliabilities increase rapidly. This table shows the need for maintaining a high level of reliability, and a corresponding low average failure rate among the subsystems.

It should be noted that the subsystem reliability range of 0.8905 to 0.9983 which corresponds to the respective overall system reliability range of 0.50 to 0.99 applies only to each composite subsystem, and not to the components and piece parts within any subsystem. While the overall system has been classified into seven major subsystems (for convenience in handling the reliability study) each of these subsystems, excluding the structure, will contain a multitude of components and piece parts which must have much higher individual reliabilities, and corresponding failure rates many orders of magnitude lower than those of the subsystems.

An estimate of the complexity of the SMS system has placed the number of individual components at approximately 10,000. On this basis, Table 7-5 was formulated to show the average component failure rate needed to maintain specified levels of overall system reliability.

2. Sensor System Reliability Guides

Of all the SMS subsystems, the sensors should be regarded as the key items in the overall system. The meteorological data will be obtained through the sensors and converted into electrical signals which will be processed for transmission to the command and data acquisition stations on Earth. The performance

TABLE 7-5
AVERAGE COMPONENT vs OVERALL SMS SYSTEM RELIABILITIES AND FAILURE RATES FOR A 10,000 COMPONENT SYSTEM AND ONE YEAR (8760 HOURS) OPERATION

Overall SMS System		Average Component	
Reliability	Failure Rate $\frac{\text{Failures}}{\text{hr}}$	Failure Rate $\frac{\text{Failures}}{\text{hr}}$	Reliability
0.99	1.14×10^{-6}	1.14×10^{-10}	0.999999
0.95	5.71×10^{-6}	5.71×10^{-10}	0.999995
0.90	11.42×10^{-6}	11.42×10^{-10}	0.999990
0.85	18.27×10^{-6}	18.27×10^{-10}	0.999984
0.80	25.11×10^{-6}	25.11×10^{-10}	0.999978
0.75	33.11×10^{-6}	33.11×10^{-10}	0.999971
0.70	40.81×10^{-6}	40.81×10^{-10}	0.999964
0.65	49.09×10^{-6}	49.09×10^{-10}	0.999957
0.60	58.22×10^{-6}	58.22×10^{-10}	0.999949
0.50	78.77×10^{-6}	78.77×10^{-10}	0.999931

of the sensors imposes a basic limitation on the quality and quantity of the data recovered.

Essentially, this performance may be expressed in terms of the following parameters:

(1) Resolution

(2) Ground Coverage Area

(3) Sensitivity

(4) Dynamic Range

(5) Storage Capability

The choice of the sensors and their associated optical and imaging systems is primarily a matter for design. However, the problems of sustaining acceptable performance under the hazards presented by the space environment are matters pertaining to reliability.

The following hazards to sensor system life and performance must be evaluated and appropriate protection should be provided in the design of the SMS system:

(1) Radiation - electromagnetic, high energy particles, and thermal from solar, Earth, and cosmic sources

(2) Extreme satellite temperatures

(3) Micrometeorite abrasion

Because many of the sensors which will be used in the SMS system to detect various types of radiation are very sensitive devices, they will be damaged if they receive excessive radiation inputs. In the case of thin film type infrared detectors, permanent damage will result if they accidentally view either the Sun, highly illuminated areas of the Earth, or its cloud cover. The length of exposure which will result in damage to these detectors varies with illumination intensity but is in the order of microseconds. This undesirable feature presents a serious hazard to the survival of an unprotected sensor system, in that the nature of the SMS mission requires the sensors to scan the entire Earth disc at any time upon command.

While the location of the Sun with respect to the Earth and satellite could be determined in advance so that the attitude control system could keep the sensors pointed away from the Sun, the location of excessively illuminated spots on Earth cannot be predicted in advance. Consequently, any sensor protective system must be designed so as to function independently of the other SMS subsystems. Furthermore, there might be times when the sensors scan the Earth under conditions where the Sun appears at the edge of the Earth's disc.

The most promising type of sensor protective system appears to be one in which the sensors are protected by a graduated density sequence of filters, such that the net transmission can be gradually increased to a safe value.

The necessity of protecting the sensors by means of filters presents a threat to achieving reliability in resolution. Because the sensors must be shielded from excessively intense illumination, a failure in the protective system which prevented the removal of the denser filters (see Volume 3) would degrade their resolution and contrast capabilities.

The sensor system, and other electronic equipment as well, should be protected against damage from high energy particles and micrometeorite abrasion. These and excessive ultraviolet radiation can produce browning, and reduced resolution in the optical equipment.

Excessive and uncontrolled satellite temperatures can reduce the resolution of infrared detectors because they produce unpredictable shifts in sensitivity. In view of the one year operational requirement, the use of cryogenic cooling would be prohibitive.

3. Power Supply Reliability

a. Electrical Power Generation Methods

A survey of the following five types of electrical power generation systems was conducted during the SMS design study:

(1) Silicon Solar Cell Arrays

(2) Solar Powered Thermionic Systems

(3) Nuclear Power Systems

(4) Chemical Power Systems

(5) Microwave Beam Power Transmission Links

Of these five system types, the most suitable for the SMS system, in terms of reliability, would be the silicon solar cell array. Within the foreseeable state of the art, there is no close second choice, because each of the other types have either serious reliability limitations, or are impractical because of size and/or weight.

The nuclear power system (SNAP) might be a favorable choice for a satellite in which a high power capability was required, if its complexity, weight, and radiation hazard potential could be brought within reasonable limits. However, it does not appear that such a development will occur within the schedule limits of the SMS.

The choice of the silicon solar cell array as the primary source of electrical power for the SMS was made on the basis of the following reliability advantages:

(1) Solar cell arrays provide direct sunlight to electrical energy conversion with no moving parts.

(2) Sunlight will be available continuously, except for a small portion (approximately 6%) of the satellite's orbit when the Sun will be occulted by the Earth.

(3) Solar cells can be arranged into modular circuit configurations which minimize the effects of individual cell failures.

(4) Solar cells do not produce any high temperatures, radiation, or other deteriorating environments which would be a hazard to other satellite systems.

However, solar cell arrays are subject to the following inherent limitations:

(1) To obtain the electrical power levels required for the SMS system, it will probably be necessary to locate the solar cells on paddles which extend from the satellite body, because of surface area limitations. Thus, to have electrical power available in orbit, these solar paddles must be successfully deployed and continuously oriented.

(2) The electrical power generation characteristics vary as a function of paddle (solar cell) temperature and illumination intensity.

(3) Silicon solar cells must be protected by a cover of glass, to avoid permanent damage caused by high energy particles radiated by the Sun.

Even with the foregoing inherent limitations, they are inherently more reliable than the other types of electrical power generation systems.

For example, the solar powered thermionic system requires focusing of the Sun's rays. Because they generate electrical power from the heating effect, they tend to complicate the satellite thermal control problem. The thermocouple type would be subject to both short and open circuit failures in the event that the operating temperature became excessive. The turbogenerator type involves moving parts which are vulnerable to failure. Unlike ordinary failures in solar cell arrays, failures in turbogenerator machinery have a high probability of totally destroying power generation capability. The microwave beam power transmission links have too short a range limitation, require extremely large transmitting and receiving antennas which must be precisely oriented, and have very low efficiencies, although they have a theoretically indefinite life.

b. Solar Cell Arrays

Two basic parameters govern the circuit configurations of solar cell arrays. The required output voltage establishes the number of cells which must be connected in series, and the load current capability requirement establishes the number of parallel banks of series strings which must be provided.

Even though solar cells may be interconnected into series-parallel groups, the output voltage and current requirements still control the overall numbers of series and parallel cells.

Apparently, the most prominent catastrophic failure mode among solar cells is that in which they fail in an open circuit condition. On this basis, selecting a low array output voltage would tend to improve reliability, because the number of cells in series would be reduced. Although the open circuit failure mode may be dominant, there is still a possibility of short circuit failures. The effect of a short circuit failure across any single cell in the circuit configuration shown in Figure 7-5A would be to reduce the total voltage in its series string. Because the output terminal voltage must be unique, such a situation would cause a circulation of current from the highest voltage string into all other strings having a lower net voltage. This circulating current could produce secondary failures in the unshorted cells, by causing excessive heating. Damage to surviving cells could be prevented by placing a diode in series with each string. However, these diodes would result in holding the array output voltage at the level of the highest string voltage, since all other strings would be reverse biased. This would result in the load current being drawn only from the highest voltage string. Such a condition would be only transient, because of the solar cell characteristics shown in Figure 7-6. As the current load on a solar cell is increased, its output voltage decreases. Therefore, the voltage of the highest voltage string would be drawn down to an equilibrium level at which other cell strings would begin to contribute load current.

The preceeding analysis assumes that the short circuit failure occurs across a cell, and not from any cell terminal to ground. For the case of a short circuit failure to ground, protective diodes in series with each string are necessary to prevent short circuiting the entire array. With protective diodes in series with each string of cells (Figure 7-5A) either a short circuit, or an open circuit failure results in the loss of the power produced by the entire string. The circuit configuration shown in Figure 7-5B shows the same number of solar cells as in Figure 7-5A, and has the same nominal output voltage and current capability. However, the cells in this circuit are completely interconnected. An open circuit failure in any number of cells, other than in a complete parallel bank pattern, does not change the output voltage of the array. It produces negligible effect on total

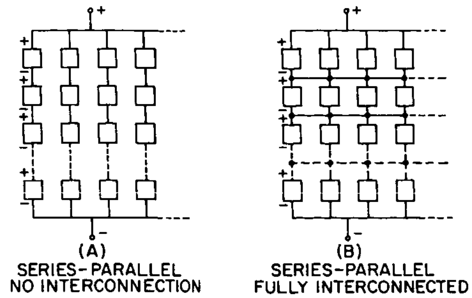


Figure 7-5. Solar Cell Array Circuit Configurations

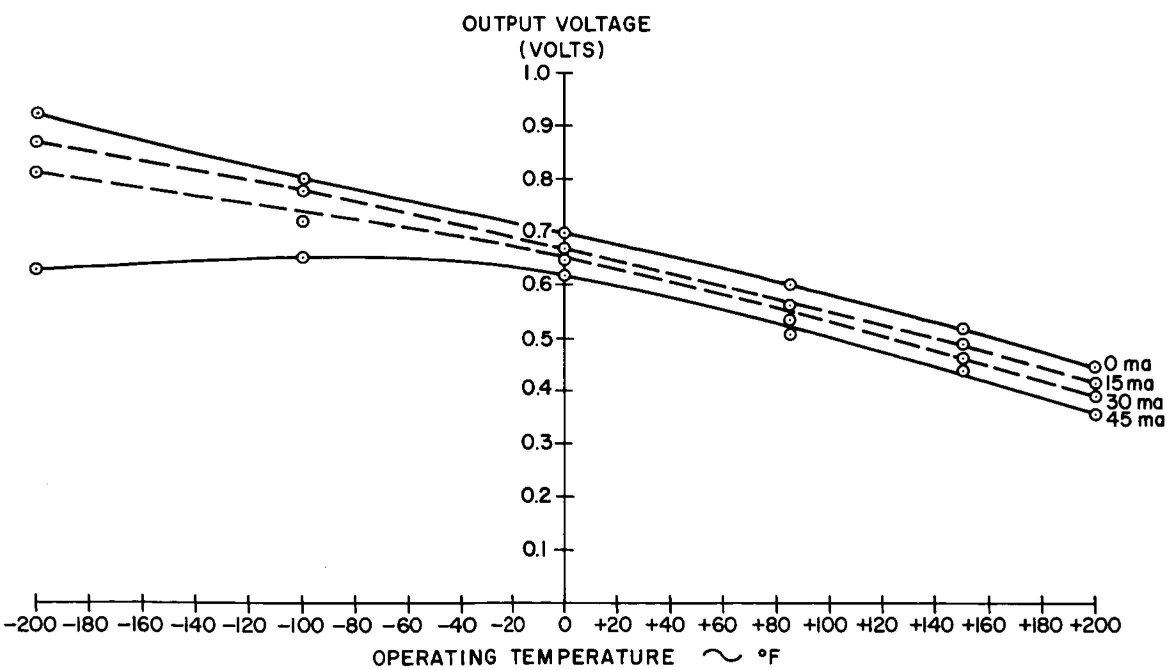


Figure 7-6. Output Voltage vs Operating Temperature for Solar Cell

load current capability for failure patterns where the open circuited cells are substantially distributed uniformly among the series layers of paralleled cells. A short circuit across a single cell will reduce the array voltage by an amount equal to that produced by a single cell. Since all of the other cells in parallel with the shorted one will be supplying current the short circuited cell, the short circuit failure may be converted into an open circuit. However, there is a limit to the number of solar cells which may be connected in parallel. This limit is set by the individual cell characteristic variations. The cells in each parallel bank must be matched in output voltage for the entire range of load currents and operating temperatures encountered. Otherwise, there will be circulatory currents which may heat and damage the array.

The circuit shown in Figure 7-5C indicates the preferred type of solar cell arrangement. This circuit is a combination of the types shown in Figure 7-5A and B. The number of cells in each parallel interconnected group is determined by the individual cell characteristics and how close the cells can be matched.

c. Storage Batteries

It will be necessary to have rechargeable storage batteries in the SMS power supply to provide electrical power until the solar cell array is deployed, oriented, and generating power, and when the satellite is occulted by the Earth. The most critical need for battery reliability comes early in the mission, prior to the time when electrical power is available from the solar cell array. The critical nature of this period stems from the need of electrical power to position the solar paddles and stabilize the satellites. If a battery failure deprives the satellite of power prior to the time when the solar cell array can generate power, the solar cell array would never get the opportunity to do so. The satellite would have failed completely.

In the event that battery power fails during an occult period, some data gathering capability will be lost until the solar array is again able to generate sufficient power. It is doubtful that during an occult period without power, the satellite will have become so destabilized as to orient the solar cell array away from the Sun so that power will not be generated upon emerging into sunlight.

In general the reliability of a storage battery subsystem can be improved by applying either standby or active redundancy techniques. For battery systems using standby redundancy, one battery is normally kept in an operating mode, supplying load power and undergoing recharging from the solar cell array. Other batteries in the system are kept in a standby status, under a trickle (low rate) change. They do not supply load power until either a malfunction or the state of charge in the operating battery requires its replacement. When such a condition occurs, the operating battery is placed in the standby status, and one of the standby batteries becomes the operating battery.

In active redundant battery systems, all batteries continuously share the load throughout the mission, and are recharged by the solar cell array during the sunlit portions of the orbit. Essentially, this type of arrangement has the batteries connected in parallel, with appropriate circuits provided to isolate the effects of individual cell failures.

Ordinarily, a standby redundant system has greater reliability than an active, or parallel redundant system with the same number of space functional elements. However, in the case of storage battery systems, the parallel redundant configuration would be inherently more reliable. The reason for this reversal lies in the relation of battery failure rate and useful life to depth of discharge. Depth of discharge represents the amount of electrical energy removed from a battery in proportion to its total energy capacity. Test data obtained from Inland Testing Laboratories, Dayton, Ohio, and from Gulton Industries, indicates that the time history of a battery's depth of discharge is a predominant factor which controls its failure rate and useful life. This test data indicates that the failure rate increases rather rapidly with depth of discharge, rather than remaining constant, as in ordinary cases. Thus, the effect of depth of discharge makes the parallel, active redundant system a more promising choice for satellite energy storage.

d. Power Conversion, Regulation, and Distribution

The solar cell array will generate DC electrical power. The output voltage will vary as a function of cell temperature and load current. The various subsystems within the satellite will require alternating currents and DC voltages other than those obtained directly from the solar cell array.

In the interest of reliability, the number of different power conversions should be held to a minimum. While the loss of power to any essential system will result in overall system failure, and reducing the number of conversion outputs will not prevent failures within individual subsystems, limiting the variety of electrical output tolerances which have to be maintained, simultaneously, will aid in providing reliable system integration. Furthermore, if the number of different electrical power conversions provided is limited, the suppliers of the individual subsystems will have to design their equipment to a set of standard power inputs, and the power conversion portion of the power supply can be frozen early in the design phase.

Reliability in the power supply system requires not only the absence of catastrophic type failures, but also performance within specified tolerance limits. The power converters, regulators, and distribution system must operate satisfactorily with the output voltage and current produced by the solar array. As shown in Figure 7-6, these parameters will vary with temperature. Also, the total variation will increase linearly with the number of series cells. Thus, a solar cell array having a high voltage output, and, consequently, a greater number of cells in each series string, will have a larger output voltage swing than an array with a lesser number of series cells and lower output voltage. To achieve maximum reliability in the power regulation function, the power converters and regulators should operate from the lowest DC array voltage practical, using components of proven reliability. Furthermore, a reduced array voltage will tend to improve the inherent reliability of the solar cell array, because each series cell aggravates the resulting power loss if cells in the string failed in open circuit.

4. Attitude Control and Station Keeping

a. Basic Satellite Control Requirements

Two types of control systems have been studied for application to the SMS, namely 3-axis stabilization, and spin stabilized. Although the spin stabilized type of system is not inherently unreliable as a control system, it does present several hazards to the reliability of other satellite subsystems, and introduces a considerable degree of complexity, especially with regard to the sensors. Therefore, because of its adverse effect upon overall system reliability, the spin stabilized type of control system should be ruled out, leaving the 3-axis stabilization type as the recommended choice.

The satellite control required for the SMS must maintain and stabilize vehicle motion about each of three body axes, and provide a station keeping ability to maintain the longitude of the satellite subpoint within ±2°. With regard to body axes control, the SMS attitude control system will be required to remove the initial vehicle body angular rates which will be imparted during injection into orbit, to align the vehicle axes to coincide with a chosen Earth oriented set of reference axes, within a tolerance dictated by sensor pointing accuracy.

For successful operation of sensors, angular motion about each body axis must be stabilized to the accuracy required to prevent picture smear.

Satellite orientation and stabilization must be maintained by the attitude control system in spite of the action of distrubing torques. The disturbance torques which are anticipated could arise from several sources, including:

(1) Solar pressure

(2) Orbital eccentricity

(3) Gravity gradient

(4) Solar paddle deployment

(5) Earth's magnetic field

(6) Operation of satellite equipment

(7) Meteorite impact

Although these torques may be small in magnitude, compensation by the attitude control system is required because of their cumulative effect over the specified one year mission.

b. Reliability Model

Even though a detailed design of the SMS attitude control and station keeping system has not yet been formulated, a basic reliability model can be devised for the purpose of providing design guidance and showing the gross relationship between subsystem module, subsystem, and overall SMS system reliability. Because the attitude control and station keeping functions are essential to the success of the SMS mission, the loss of any functional capability constitutes a failure of the mission. Accordingly, each of the attitude control and station keeping functions can be represented as a series element in the reliability model. In turn, each of these functions can be resolved into the following three component groupings:

(1) Position sensor, error detector

(2) Servo amplifiers and networks

(3) Control torquer thruster

This arrangement results in the reliability model shown in Figure 7-7, which has a total of 12 component groups which represent the following four attitude control and station keeping functions:

(1) Pitch control

(2) Roll control

(3) Yaw control

(4) Station keeping

As shown in Figure 7-1, the entire stabilization and control subsystem reliability represented by R_A, is one of seven major subsystems in the entire SMS system. The reliability of the structure is assumed to be unity, for the reasons given previously in the overall system and subsystem reliability requirements.

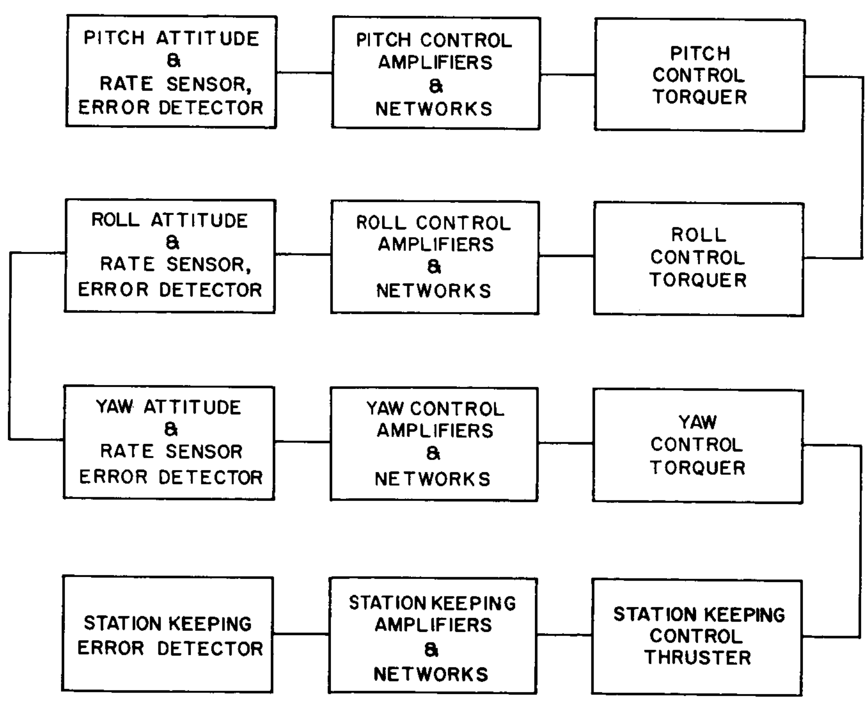


Figure 7-7. Attitude Control Reliability Model

The following table shows the reliability needed for each component grouping (R_G - Uniform), each control function (R_F - Uniform for pitch, roll, yaw, and station keeping), and the complete control system (R_A), to attain specified levels of overall SMS system reliability.

TABLE 7-5
RELIABILITY RELATIONSHIPS FOR SMS STABILIZATION AND CONTROL

Overall SMS, R_o	Stabilization and Control, R_A	Control Function, R_F	Component Group, R_G
0.99	0.9983	0.999575	0.999858
0.95	0.9915	0.997875	0.999292
0.90	0.9826	0.995650	0.998550
0.85	0.9733	0.993325	0.997775
0.80	0.9635	0.990875	0.996958
0.75	0.9532	0.988300	0.996100
0.70	0.9423	0.984575	0.994858
0.65	0.9306	0.982650	0.994217
0.50	0.8905	0.972625	0.990875

5. Communications and Data Handling

a. The Basic Earth Satellite Communications Problem

Essentially, the typical Earth satellite communications link involves a two way transmission of information between one or more Earth stations and one or more satellites. For the SMS study, only one satellite was considered. The SMS communications links must perform the following functions:

(1) Transmission of wideband sensor data from the satellite to the Earth stations.

(2) Transmission of narrowband telemetry data from the satellite to Earth stations.

(3) Transmission of command signals from Earth stations to the satellite.

(4) Retransmission of processed meteorological data signals received from Earth stations to other Earth stations (relay link).

Examining each of these required functions, in turn, it can be noted that for any given one way transmission, the criterion for success is the reception of acceptable information at the receiving station. Successful reception requires a properly functioning receiver and sufficient signal strength at its antenna. The signal strength at the receiving antenna depends upon the transmitter power, the directional pattern and orientation of both the receiving and transmitting antennas, and the attenuation of the transmission media.

There is a trade-off relationship between antenna directivity, transmitter power level, and communications reliability. A highly directive antenna concentrates the transmitted energy to produce a greater field strength at points within its beam than would be produced by an isotropic radiator. However, at points outside the beam, the signal strength produced by a directive antenna is negligible in comparison to that produced by an isotropic radiator. For transmission from a ground station to a satellite, where weight limitations do not restrict the power level, a highly directive antenna offers less reliability than a broad beam antenna (isotropic radiator), because it must be pointed accurately at the satellite. For receiving signals from a satellite, a certain amount of directivity in the transmitting antenna is needed to reduce the output power requirement. In the case of the SMS system, the transmitting antennas should be designed to provide a beamwidth sufficient to encompass the entire Earth's disc plus an additional amount sufficient to compensate for orientation errors. Thus, the reliability of an Earth-satellite communications link depends upon the reliability of the satellite attitude control, as well as upon available power and the antennas.

Because the communication links are so vital to the mission, especially the command and telemetry links which could be used to initiate corrective action in other systems, it is anticipated that it would be profitable to provide as much redundancy in the SMS communications as is permissible.

b. Data Handling Reliability Guides

A system such as the SMS will be required to receive, store, and process a variety of data in the form of electrical signals both digital and analog. Both incoming and outgoing data must be handled. The incoming data will consist of command signals and ground station processed data for relay to other ground stations. The outgoing data will consist of that which comes from the sensors and the telemetry instrumentation.

The data handling functions are, in general, all essential to a successful mission. The loss of any substantial data handling capability will constitute a failure of the mission. For a reliable data handling system, the following factors must be evaluated and adequately provided for in the system design:

(1) Compatibility among signal characteristic tolerances

(2) Adequate storage and access capability

(3) Error correction and verification

Compatibility among signal characteristic tolerances is essential to a reliable data handling system because it tends to reduce, if not eliminate, the probability of misinterpreting information signals.

Storage capacity should be sufficient to meet the demands of the sensors, any time sharing, and/or transmission delays. In addition to sufficient storage capability, in terms of a number of information bits, the information must be accessible when required. Methods of storage which do not provide sufficiently rapid access are detrimental to a reliable system.

In a digital computer, more so than in an analog computer, means for detection and correction of errors can be readily provided. For the SMS system, the incorporation of an error detection and correction system will provide improved reliability. Verification by means of retransmission to the command and data acquisition station will provide a means of detecting errors, and correcting them through appropriate commands.

6. Structural Reliability

The basic structural design problem for the SMS is to provide a high strength minimum weight vehicle structure which is capable of withstanding both the launch and the orbital environments. Also, this structure must survive ground handling without deterioration. The results of design studies indicate that the satellite will experience its most critical loading during the period between launch and when it attains final orbit. In this period, which is at the beginning of the mission, the deteriorating effects of high vacuum and radiation associated with the space environment will be negligible, because of the short period involved. In the final orbit, structural loading will be negligible, whereas the problems of material survival remain.

Insofar as the structure is concerned, it is believed that its reliability will be unity. For the case of structures which are designed properly, they are either successful or fail during the mission, independently of time. Except for damage to materials resulting from the space environment, structural reliability is fixed the moment the satellite is launched, and is relatively immune from wearout and random time failures.

Republic has conducted company sponsored studies and experiments for the purpose of developing structures and materials suitable for spacecraft. It is believed that, by the proper choice and application of materials, the probability of structural failure will be negligible.

7. Thermal Control Reliability

a. Satellite Temperature Control Requirements

A satellite such as the SMS will be exposed to thermal radiation from the Sun and Earth and, to a negligible extent, from other celestial bodies. It will simultaneously emit thermal radiation into space (an infinite heat sink having a temperature of approximately 4°K). The temperature levels and distribution pattern within the satellite and on its exterior surface are governed by its exterior geometry, structural heat paths, internal heat sources, orientation in space, absorptivity and emissivity of its exterior surface, and its thermal control system. All of the factors which govern temperature within the satellite must be controlled to have an operating satellite system. Accordingly, the SMS thermal control system is represented as a series element in the overall SMS reliability block diagram.

In a generalized parametric study, it is difficult, if not impossible, to establish a quantitative reliability value for the SMS thermal control system. At best, only an outline of general principles and application techniques can be formulated to aid in the design of a specific system.

The design of any SMS thermal control system must start with a careful analysis of the allowable temperature environments for each component in the entire satellite. In addition, the location of each component and the distribution of internal heat sources must be considered. The component allowable temperature environments will establish the basic thermal control requirements and place restrictions upon the type of control system which can be used.

b. Thermal Control Systems

The function of the SMS thermal control system will be to adjust the net heat flow from the satellite so that all components will be operating within allowable temperature environments. Three basic types of control systems are being considered for this purpose, namely:

(1) Passive systems

(2) Semi-active systems

(3) Active systems

Each of these systems has its merits and limitations insofar as performance and reliability are concerned.

The passive system is the least complex, and offers the highest reliability because it has no moving parts or fluids. In the passive system, temperature control is achieved by controlling the absorptivity and emissivity of characteristics of the satellite's exterior surface. Apart from degradation of these surface characteristics, such as may result from the space environment, the passive type control is not subjected to failure as a function of expired time. The passive thermal control system will work as well at the end of the mission as it did at the beginning, because the degree of surface degradation is negligible for the one year mission specified.

The inherent disadvantages in the passive system are that it is not flexible enough to allow for variations in satellite system configuration, and it cannot maintain a tight range of control over the individual component temperatures (or even the average satellite temperature). Once the thermal characteristics of a passive control system are fixed, the temperature distribution is fixed for a given set of heat sources and sinks.

The selection of a passive type thermal control system for the SMS poses the problem of controlling satellite temperatures during the brief period when the satellite is occulted from the Sun.

The semi-active type system offers greater flexibility in comparison to the passive type in that the net heat transfer rate can be varied to accommodate changes in satellite temperatures. This system, while it may incorporate moving parts and/or fluids, specifically excludes a heat pump. A typical example of the semi-active class of systems is the type where bimetallic temperature sensing elements are connected to operate a shutter-radiator device, wherein the net transfer of heat is controlled by varying the position of the shutter. An alternate method, within the same class of systems, would be to actuate the shutters by means of an independent actuator either through an automatic program, or upon command.

Some reduction in reliability over that provided by the completely passive system is inherent in the semi-active system, because of the necessity for moving parts. But the type of mechanical linkages used are simple and not subject to the same failure rates as are the active systems which use a heat pump.

An active control system, using pumps, coolants, radiators, and compressors represents the most flexible, and probably the most unreliable system for unattended operation. While very close control can be maintained of the units, the system depends on passive devices, such as oriented radiators, to dissipate heat. These, in turn, require controlled systems. The complexity of the system tends to reduce reliability.

APPENDIX A - ALTERNATE DESIGN CONCEPTS FOR WEIGHT SAVING

A. ALTERNATE SOLAR PADDLE ARRAY STRUCTURE

To date, solar paddles have been conceived as rigid surfaces on which solar cell modules were mounted. These surfaces were either fixed to the basic satellite body at appropriate angles, or oriented about one or more axes to face the Sun. The rigid panels posed many problems to stress and vibration amplification during the launch phase. The proposed concept is that of flexible panels that can be wrapped around a satellite body during launch and later deployed in space. Figure A-1 illustrates the concept. The paddle frame consists of two longitudinal tubular members separated by two flat lateral springs attached to their ends. The solar cells are bonded to narrow vertical slats, the ends of which are fastened to the lateral springs. A wire mesh fastened to the rear surface of the slats and to the framing members completes the configuration. During launch, the paddles are supported by the rotating shaft at the inboard tubular members and by two pyrotechnic pin pullers at the ends of the outboard members. An alternate support scheme could employ an arrangement of Marman clamps previously described. When the vehicle to which the paddles are attached has attained orbit, the lateral springs provide the necessary power to deploy the paddles. A problem associated with this scheme is the need of cancelling out the oscillations arising from deployment. In the absence of an atmosphere, structural damping would have to be relied upon. One method for effecting damping would be to fabricate the springs as a laminate with an interlayer material providing good friction damping.

The second scheme visualized for construction of the paddle frame is shown in Figure A-2. In this concept the rigid solar cells would be attached to vertical slats and a wire mesh backup as in the previous scheme. The framing member, however, would be an inflatable toroid ring structure of circular cross section, constructed of a material capable of being rigidized upon deployment and exposure to the space environment. Rigidization of the material would impart sufficient stiffness to the toroid to hold the wire mesh taut and allow the overall paddle assembly to be rotated relative to the spacecraft body without inducing appreciable torsional distortions or wind-up of the solar cell backup. Several different methods of construction for obtaining structural rigidization have been discussed in Section 5. In the deflated condition, the tube and cell backup assembly would be stowed against the spacecraft shell and secured by two Marman clamps. The inner surface of the clamps and adjacent areas of the shell would have segmented flexible foam blocks bonded on to prevent damage to the frame materials. The entire system would be deployed by application of sufficient gas pressure. The pressure would also serve to hold the tube in its final shape until rigidizing has been completed. A short rigid end segment would be provided at one section of the assembly for adaptation of a rotating hinge arm to permit rotation of the paddle.

The primary structural problem attendant to both of the paddle framing schemes described is that of providing adequate support for the solar cells when they are in the stowed position to enable them to survive all of the anticipated vibratory loads. One method that has been considered for solving this problem is that of installing an air cushion between the spacecraft and the solar paddles . The cushion would be bonded to the paddle. Such cushions would be fabricated from a photolyzable material designed to volatilize in the space environment. Another possible solution for circumventing the vibration problem is the use of flexible thin-film photovoltaic cells now under development (see Volume 5). The cells are fabricated by the evaporation of

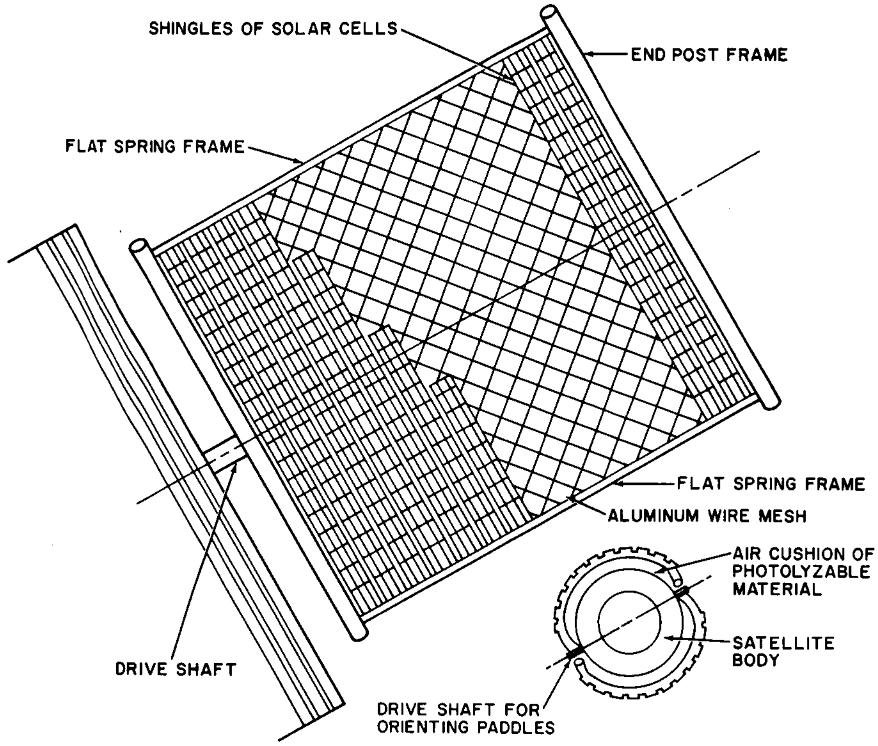


Figure A-1. Flexible Solar Panel

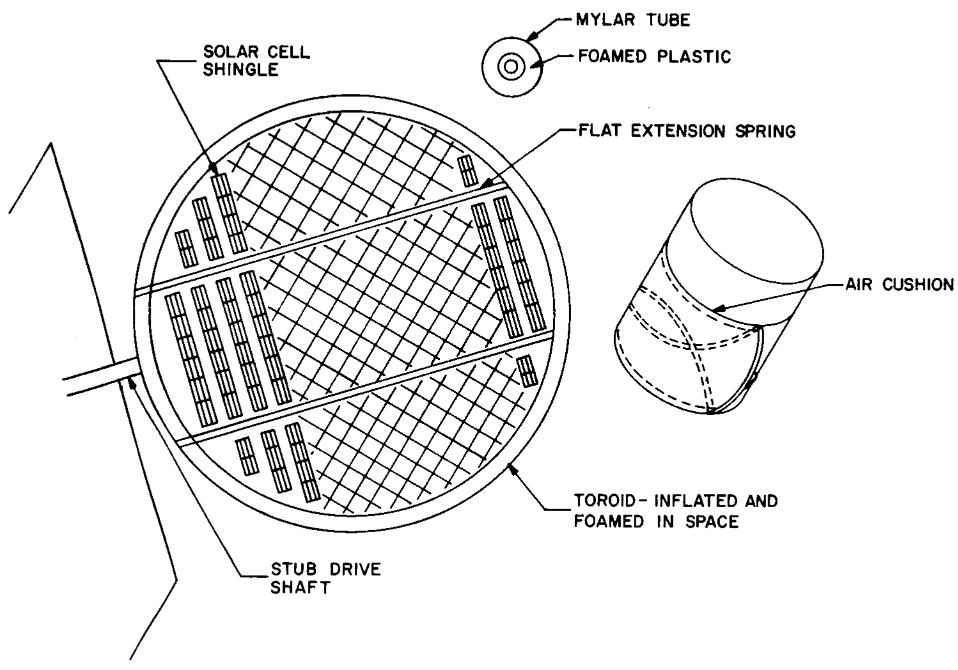


Figure A-2. Inflatable Solar Panel

thin layers of semiconductors onto thin flexible substrates of Mylar or other suitable material. A typical cell would be 6 in. square and about 0.008 in. thick. Besides simplifying the problem of storage, the use of these cells would also provide an additional weight saving over conventional silicon cells. However, the present low efficiency of the cells requires the use of more cells, with an attendant substantial increase in area over that required for the silicon cells.

In addition to the problem of launch vibration, other problems requiring solution for development of a practical solar paddle system consists of:

(1) Prevention of rupture of the wire mesh upon deployment of the paddle prior to the onset of tube rigidization.

(2) Imparting sufficient pressure to the tube to tauten the wire mesh prior to rigidizing.

(3) Provision of adequate thermal control to preclude intolerable thermal distortions of any segment of the system.

(4) Provision of adequate pressure control to prevent "under pressurization" during the critical rigidizing stage.

(5) Selecting the best films, catalyst systems, and fabrication methods to minimize weight.

The estimated weights for the panels described are shown in Table A-1. Figure A-3 illustrates the projected trend of solar panel weights.

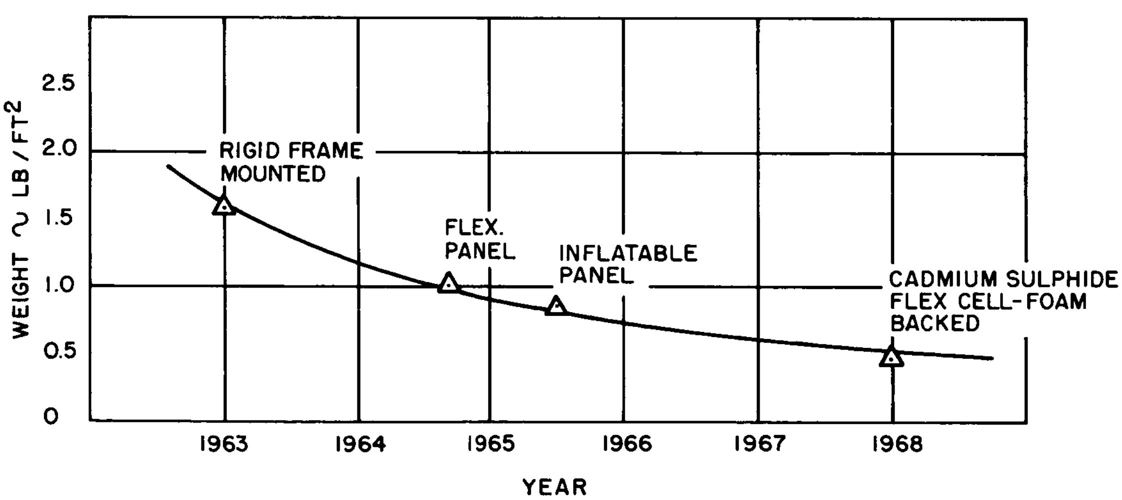


Figure A-3. Projected Weight Trend of Solar Panels

TABLE A-1
WEIGHT COMPARISON - FLEXIBLE SOLAR PANELS

Type	Honeycomb Frame/Si/Cell*	Metallic Frame Si Cells	Metallic Frame Cd/s Cells	Foam Frame Cd/s Cells
Cell Area (ft^2)	25	25	75	75
Cell Weight (lb)	17.51	17.51	10.50	10.50
Frame Weight (lb)				
Mesh		2.00	6.00	6.00
Metal	4.00	6.34	5.59	1.30
Foam				2.04
Mylar		1.05	1.05	1.05
Honeycomb	13.46			
Total (lb)	34.97	26.90	23.14	20.89
Panel wt (lb/ft^2)	1.39	1.08	0.31	0.28

* Standard construction - not flexible

B. SUBLIMING SOLID FOR ATTITUDE CONTROL GAS GENERATION

A significant saving can be realized by a substitution of gas supply in the attitude control system. As presently proposed, nitrogen at 3000 psi will be stored as a gas source for dumping momentum in the attitude control wheels and for station keeping. As finally used for reaction control, the gas pressure is reduced to about 6 psi. Because of the high pressure this system requires heavy storage tanks and multiple pressure reducing valves and weighs as much as the gas itself.

Rocket Research Corporation has produced a reaction jet system that generates gas by subliming a solid. The gas is generated and stored at approximately one-half atmosphere, thereby allowing the weight of the storage tank to be held to a nominal figure, and eliminating the need for pressure reducing valves.

The specific impulse of the generated gas is comparable to that of stored nitrogen (see Figure A-4), and hence, fuel, or gas weights are similar. Table A-2 compares the weights of the high pressure nitrogen system and the subliming solid system. A more complete description of this system may be found in Volume 4.

C. EXPLOSIVE STAGE SEPARATION

The weight of the separation mechanism may also be reduced by application of pyrotechnic technology. An explosive linear shaped charge (LSC) can be used to supplant the Marman clamp, matched machined rings, and explosive bolts used to separate the Marman clamps presently being considered. It would be necessary to mechanically fasten the present adapter to the launch vehicle spin table, and then part the adapter at the proper time with the LSC. A system similar to this is now employed on OGO. It is estimated that a minimum of 8 lb could be saved on the medium capability satellite by use of the LSC system.

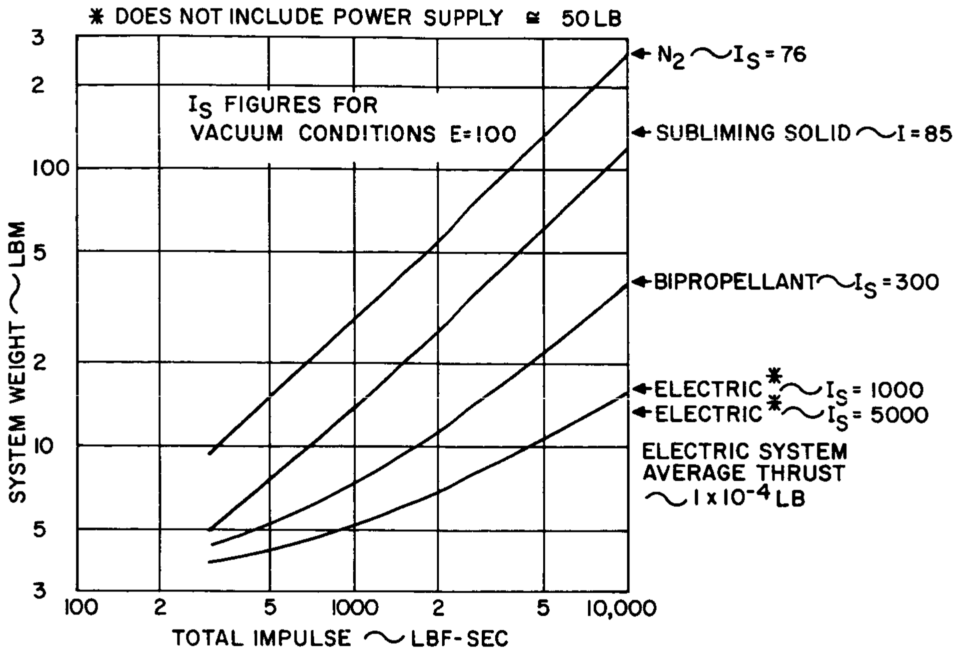


Figure A-4. Attitude Control System Weights

TABLE A-2
WEIGHT COMPARISON - 3000 PSI
NITROGEN SYSTEM vs SUBLIMING SOLID SYSTEM

Item	Nitrogen Gas System (lb)	Subliming Solid (lb)*
Propellant	24.00	16.50
Tanks and Valves	34.00	0.80
Jets (7 at 0.25 lb ea)	1.75	1.75
Manifold	---	0.10
Lines	0.25	0.15
Total	60.00	19.30

* I_{sp} = 1406 lb-sec